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SP-100 PROJECT
Nuclear Reactor Power for a Space-Based Radar

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Prepared by the Jet Propulsion Laboratory, California Institute of Technology, for the U.S. Department of Energy, the U.S. Department of Defense, and the National Aeronautics and Space Administration.

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ABSTRACT

A space-based radar mission and spacecraft, using a 300 kWe nuclear reactor power system, has been examined, with emphasis on aspects affecting the power system. The radar antenna is a horizontal planar array, 32 x 64 m. The orbit is at 61 deg, 1,088 km.

The mass of the antenna with support structure is 42,000 kg; of the nuclear reactor power system, 8,300 kg; of the whole spacecraft about 51,000 kg, necessitating multiple launches and orbital assembly. The assembly orbit is at 57 deg, 400 km, high enough to provide the orbital lifetime needed for orbital assembly.

The selected scenario uses six Shuttle launches to bring the spacecraft and a Centaur G upper-stage vehicle to assembly orbit. After assembly, the Centaur places the spacecraft in operational orbit, where it is deployed on radio command, the power system started, and the spacecraft becomes operational. Electric propulsion is an alternative and allows deployment in assembly orbit, but introduces a question of nuclear safety.

ACKNOWLEDGMENT

William Gray, John Winslow, and John Spanos of the Jet Propulsion Laboratory and Joan Boudreau of Los Alamos National Laboratory contributed significantly to work described in this report. Gerry Tsandoulas and Darryl Weidler of Lincoln Laboratory, Massachusetts Institute of Technology, were most helpful in discussions of radar aspects.

The work described herein was conducted by the Jet Propulsion Laboratory, California Institute of Technology, by the Los Alamos National Laboratory, University of California, and by the NASA Lewis Research Center, for the U.S. Department of Defense, the U.S. Department of Energy, and the National Aeronautics and Space Administration.

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EXECUTIVE SUMMARY

This study examines the use of a 300 kWe (kilowatts electric) nuclear reactor power system for a Space-Based Radar (SBR) that observes moving objects. Aspects of the mission and spacecraft bearing on the power system were also considered.

Important mission and spacecraft requirements were that the spacecraft will use 300 kWe of prime power; the radar antenna will be a horizontal planar array, 32 x 64 m; the antenna shall have an unobstructed downward view; preferably, no spacecraft elements shall extend beyond the antenna rectangle; the dose of ionizing radiation from the power system to the antenna shall not exceed 1×10^5 rad, integrated over five years of operation; the orbit shall be at 61 deg inclination, approximately 1,100 km altitude; and the pointing accuracy shall be plus or minus 0.2 deg.

The spacecraft mass, excluding propulsion, was found to be about 51,000 kg, of which 8,300 kg is the mass of the power system. The energy source is a fast-spectrum reactor fueled with uranium nitride and cooled with liquid lithium. A shield shadows the rest of the spacecraft from reactor radiation and an extendable boom further reduces the dose. Pumped lithium heats one end of a set of Si-Ge thermoelectric elements. Waste heat from the cold end of the thermoelectrics is removed by heat pipes and radiated to space. Electrical power produced by the thermoelectrics is conditioned and delivered to the rest of the spacecraft as constant voltage dc. The reactor operates at constant power and temperature. Load changes at rates up to 3-30 kWe/s are handled by dumping unneeded power through shunt resistors in the power system; faster changes are handled by capacitors in the radar antenna system.

The power system boom and main radiator fold to permit the system to fit within a 9-m-long portion of the Shuttle cargo bay or Titan 4 launch vehicle fairing. They deploy on command. The deployed length of the power system is about 25 m; the width, about 20 m. The thermal radiation from the power system to the rest of the spacecraft is limited to 1 sun.

The radar antenna, with its supports and interconnects, has a mass of 42,000 kg. It is divided into four 16 x 32 m quadrants. Each is stiffened by trusses. Additional structure connects the quadrants to each other and to the rest of the spacecraft.

Communications are relayed through the Tracking and Data Relay Satellite System (TDRSS) or another satellite. Small medium-gain and low-gain communications antennas are provided on the SBR spacecraft. Navigation is by Global Positioning System (GPS).

Attitude control sensors include horizon and celestial sensors. Control torque is provided by control moment gyros, which are unloaded by interaction with the Earth's magnetic field. Attitude changes will be relatively slow, but not because of the power system.

Two spacecraft configurations were evaluated, one with the power system boom parallel to the radar antenna and one with it perpendicular. The parallel configuration led to problems with undesirable radar reflections from elements of the spacecraft and with an asymmetrical mass distribution affecting the attitude control. The configuration with boom perpendicular to the radar antenna was, therefore, selected.

Various methods of placing the spacecraft in its operational orbit were considered. Preferred is a series of six Shuttle launches from Cape Kennedy, which place in an assembly orbit at 57 deg inclination, about 400 km altitude, four quadrants of the radar antenna, the remainder of the spacecraft, and an off-loaded Centaur G-class upper-stage booster vehicle. (This work was done before the NASA decision not to carry Centaur on the Shuttle.) The elements brought up in each Shuttle launch are left in orbit with passive temperature control and no attitude control. Each successive Shuttle flight recovers the parked package and, using the Remote Manipulator System (RMS) and Extra Vehicular Activity (EVA), assembles it to the element just brought up, and releases it to await the next flight. The time needed for the multiple Shuttle flights, which may be as much as a year, means that the parked assemblies must remain in orbit for this length of time. A minimum altitude of about 400 km is needed to assure such orbital lifetime. After the last Shuttle flight, the Centaur brings the spacecraft to operational orbit where its boom, main radiator, and radar antenna are deployed.

An alternative scenario uses an electric propulsion module in place of the upper stage. This module has ammonia propellant and arcjet thrusters. The boom and antenna are deployed in the assembly orbit, permitting considerable checkout before the Shuttle leaves. The power system is then started and provides power for propulsion to the operational orbit. This scenario requires careful examination of possible safety problems associated with starting the reactor at 400 to 500 km altitude. In case of malfunction, the orbital lifetime of the spacecraft and, therefore, the time for radioactivity to decay before re-entry, is rather short at such altitudes.

If Titan 4's are used for launch in place of the Shuttle, one or more Shuttle flights could be added for astronaut-aided assembly of the elements parked in orbit by the Titans.

Assembly at the Space Station is possible but, because the inclination of the Station orbit (28.5 deg) differs considerably from that needed by the spacecraft, the required propulsive energy is very high. Suggested is electric propulsion using xenon propellant with ion thrusters. This does not require additional launches, but the orbital transfer time is about 15 months.

Safety was given first priority in the power system design and in operational plans. Except for zero-power testing, the reactor is not turned on until a stable orbit is reached. Until then, it will not contain significant radiation inventory. The reactor is designed to remain subcritical after any credible handling, launch, or ascent accident, and during re-entry, ground impact, and immersion in water, or burial in soil. It is never operated near the Shuttle or the Space Station. Mission profiles and orbital lifetimes ensure that, even if a failure occurs in flight, the probability of hazardous exposure from the reactor is very low. Once the reactor has operated, it is advisable to turn it off and allow time in orbit for most of the radioactivity to decay to minimize the risk associated with re-entry. At the end of mission, the reactor is turned off by ground command, backed up by an on-board clock. (The SP-100 Project policy of placing the power system in a permanent storage orbit at end of life was adopted too late to be reflected in this report.) If electric power for control is lost, spring-driven safety rods or drums will shut down the reactor. If communications are lost for more than a preset interval, the control system will shut down the reactor. Two independent shutdown means are provided. The reactor will remain intact during re-entry and will not scatter residual radioactivity.

Some of the findings concerning the nuclear power system are:

(1) Extended orbital storage of the power system will be needed for missions involving multiple launches and orbital assembly. It seems that this will be possible without attitude control, using only passive temperature control.

(2) The shield prescribed by the current space reactor power system specification is not adequate from the space-based radar that was studied. The shield configuration, both thickness and area, will have to be specific to the spacecraft on which the power system is used.

(3) The currently specified limit on thermal radiation from the power system to the rest of the spacecraft may be difficult to meet with some radiator configurations, and may be important in selection and design of the radiators. Also, the current limit may be too high for the space-based radar and, perhaps, for some other spacecraft.

(4) Ability to operate the reactor at low power levels appears unnecessary for the mission examined in this study.

Some findings relevant to the space-based radar mission are:

(1) Because of the large mass and size of the radar antenna, multiple Shuttle or Titan 4 launches are needed to put the spacecraft into orbit.

(2) Extensive orbital assembly will be required. Shuttle-based techniques for such assembly will have to be defined and developed.

(3) Assembly at the Space Station appears undesirable because of the large difference in inclination between the Space Station orbit and the spacecraft operational orbit, requiring a very large orbital velocity increment for transfer between these orbits.

(4) Extended orbital storage will be needed during orbital assembly. Temperature control solutions are needed for each of the spacecraft elements parked in orbit during the assembly sequence.

(5) If start-up of the power system in assembly orbit or other low Earth orbit is contemplated, nuclear safety design, as it pertains to planned or unplanned re-entry, will be very important.

(6) Because of the low structural frequencies expected, rapid changes in spacecraft attitude will probably not be possible.

(7) It is not clear whether electronic components on the radar antenna can be kept within permissible temperature limits when the antenna is receiving the currently allowable thermal radiation equivalent to 1 sun from the power system, plus thermal radiation from the sun and Earth.

(8) Launch of the radar antenna selected for this study will be very costly and requires a major commitment of launch resources. There is much incentive to reduce the antenna mass and to find mission profiles with fewer launches.

SECTION I

INTRODUCTION

This report presents results of a study of the use of a nuclear reactor power system for a space-based radar. Aspects of the mission and spacecraft that bear most directly on the power system are also considered.

The report was prepared by the Systems Design Audit Team of the SP-100 Project. The goal of the SP-100 Project is to develop and demonstrate a multi-hundred kilowatt Space Reactor Power System (SRPS).

Among the objectives of the Systems Design Audit Team are to define system requirements for a 300 kWe SRPS through examination of candidate missions and spacecraft, and to examine problems of SRPS utilization for selected missions. Its approach for fiscal year 1986 (FY'86) included examination of two candidate missions to assess their implications for the SRPS, and to throw light on the use of SRPS for such missions. The missions selected were a Space-Based Radar (SBR) and an Orbital Transfer Vehicle (OTV). Work on the OTV is described in Reference 1; work on the SBR is the subject of this report.

SECTION II

MISSION SELECTION AND REQUIREMENTS

A. SELECTION AND REQUIREMENTS

Prior to the initiation of this study, 300 kWe had been selected as the design power level for development and ground test of key portions of a SRPS. A power level of 300 kWe was assumed for the spacecraft to maximize applicability of the study to the planned SP-100 effort.

A radar mission was chosen for study because of potential user interest and potential appropriateness of the 300 kWe power level. A phased array radar to observe moving objects was selected rather than a side-looking radar to observe the Earth's surface, because more information about possible missions and spacecraft was available, and interest among possible users was better established.

Mission requirements were established after discussions with G. Tsandoulas and D. Weidler at Lincoln Laboratory, and R. Jordan at JPL (Appendix B-1). They are:

Operational orbit:	Approximately 1,100 km altitude circular at 61 deg inclination
Operating life:	5 years
Radar antenna size:	32 x 64 m
Field of view:	Radar antenna shall have an unobstructed view over 2 pi steradians, centered on the nadir
Deployment:	Assembly on-orbit is permissible
Prime power level:	300 kWe
Duty cycle:	10-100%
Power output form:	To be determined (TBD)
Permissible dose of ionizing radiation to antenna	
From SRPS:	1×10^5 rad
From all sources:	1×10^7 rad
Initial operational capability	Year 1998

B. DISCUSSION OF REQUIREMENTS

The 300 kWe of prime power was assumed as given, not derived from quantitative analysis of radar needs for a specific mission. However, the performance of the radar would be enhanced by using 300 kWe rather than lower power. For instance, objects with lower radar cross-section could be detected and detection could be ensured for a shorter travel distance of the objects.

Also, the 32 x 64 m antenna size was not derived from detailed examination of radar performance needs. Rather, a previous study by Lincoln Lab had indicated that a 16 x 32 m antenna was desirable for a specific mission at roughly 75 kWe power; Lincoln Lab personnel suggested that an antenna with four times this area would be appropriate for four times the power. The larger antenna would reduce the clutter and permit detection and tracking of objects with lower radar cross-section (Appendix B-1).

The prime power is to be available at all times that the radar is in operational status. Thus, the radar could be used during any or all portions of its orbit. This contrasts with a solar-powered orbital radar, for which eclipses by the Earth generally limit availability of the power to a portion of each orbit. Batteries must then be used to store the energy for the radar. The available power, averaged over the orbit, will be less than that from the nuclear power system with the same peak input power. For the same peak power from the primary source, same radar power, and same number of spacecraft, a larger portion of the Earth's surface can be covered with nuclear-powered radars than with solar-powered. Also, the ability to operate the radar continuously provides an option to reduce on-off cycling of the Transmit/Receive (T/R) modules and the problem of temperature fluctuations during eclipse could be ameliorated.

Performance of several space-based radars utilizing SP-100 power sources are discussed in Reference 2. Performance of space-based radars has been examined by R. Jones.[1]

The radar antenna T/R modules are assumed to use GaAs components. Hence, the high total radiation dose allowable. Most of this is, however, reserved for natural and hostile radiation.[2] The spacecraft orbit is in a region of high electron flux due to the Earth's inner radiation belt. Through thin shielding, the resulting dose can be very high.[3]

C. OPERATIONAL ORBIT

The operational orbit was specified as "approximately 1,100 km" altitude circular at 61 deg inclination. Earlier orbit analysis[4] showed that at an altitude of 1,088 km, nominal eccentricity about 0.001, there is an orbit which reduces the excursion in altitude due to harmonics in the Earth's gravity field without the need for orbit circularization maneuvers. This orbit was selected.

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[1]Jones, R., SP-100 SBR Study, Jet Propulsion Laboratory, Pasadena, California (in final review).

[2]Jones, R., Radiation Tolerances for SBR T/R Modules, JPL IOM 312/84.3-2806 to L. Jaffe, December 10, 1984.

[3]See Appendix pp. 10-15 of Footnote 1: Horton, C., Natural Environment Definition for SP-100/Mission A.

[4]See Appendix pp. 1-7 of Footnote 1: Uphoff, C., SBR Orbit Analysis.

SECTION III

SPACECRAFT AND POWER SYSTEM FUNCTIONAL REQUIREMENTS

A. REQUIREMENTS

Using the previously mentioned mission requirements and other inputs from Lincoln Laboratory (Appendix B-1), functional requirements were established as follows.

1. General

The Technical Specification for the SP-100 Space Reactor Power System (SRPS): Exhibit 1 (Reference 3) is the system specification that shall be used as a guide.

2. Spacecraft Orientation

The spacecraft shall operate with the radar antenna plane horizontal and the long axis of the antenna parallel to the orbital velocity vector.

3. Edge Reflections

No portion of the spacecraft outside the radar antenna rectangle shall be within 1 m of the plane of the nadir face of the antenna.

Preferably, no portion of the spacecraft shall be outside the radar antenna rectangle.

4. Pointing Requirements

The pointing angle accuracy during SBR operations shall be plus or minus 0.2 degrees.

The spacecraft yaw angle (angle about the nadir axis) shall vary plus or minus 3.5 degree/orbit, synchronized with the sine of the latitude.

5. Power Requirements

During operation the SRPS shall be capable of delivering 300 kWe to the rest of the spacecraft. This includes transmitter and all other loads.

The SPRS shall maintain voltage as prescribed in the systems specification while following load changes of 0-100% in (TBD) milliseconds.

The interface for power output from SRPS to the rest of the spacecraft shall be per the system specification.

Additional power processing and distribution required by the transmitter and other spacecraft systems shall be provided by those systems.

6. Maximum Acceleration

After deployment of the antenna and the radiator, maximum acceleration shall be 0.01 g parallel to the boom. Acceleration perpendicular to the boom is not required.

7. Structural Requirements

The lowest frequency of free vibration of the spacecraft shall be no less than 0.01 Hz.

The effective shape of the antenna shall be maintained flat within 10 mm (0.4 in.) by structural or electrical means.

8. Survivability

Survivability requirements shall be as stated in the system specification.

B. DISCUSSION OF REQUIREMENTS

The Technical Specification for the SP-100 Space Reactor Power System (SRPS): Exhibit 1 cited (see Reference 3) is not the current version, but the system specification in effect at the beginning of this study. As a result of this and other studies, an update reflecting conclusions of the studies has been issued (Reference 4).

Fine-pointing and scanning of the radar beam is done by phasing T/R modules distributed over the antenna.

The rate of load-following was examined during the study, as noted in Appendix B-2.

SECTION IV

SPACECRAFT SYSTEMS

Among the major spacecraft systems are the radar, SRPS, attitude control, command, data handling, and communications.

A. RADAR SYSTEM

The radar antenna is divided into four quadrants, each 16 x 32 m. Each quadrant includes T/R modules distributed over the upper side of the antenna. The quadrant is stiffened and held flat by trusses above the T/R modules. This structure is sized to keep the antenna flat as required during operation, but will also support the antenna (not operating) at axial acceleration of 0.1 g. Additional structural members interconnect the quadrants in the operational configuration and connect them to the central body of the spacecraft. In operation, the long (64 m) axis of the antenna is parallel to the orbital velocity vector.

The radar system incorporates power processing specific to the radar. Capacitance is included which permits the peak transmitted power to exceed the 300 kW average power provided by the SRPS and provides short-period load following of the radar output.

The performance of the radar system is classified and is not described here.

B. SPACE REACTOR POWER SYSTEM (SRPS)

The SRPS is described in Section V. Once brought up to power, it can provide 300 kW dc to the rest of the spacecraft.

C. ATTITUDE CONTROL

The attitude control system is described in Section XI.

D. COMMUNICATIONS

Small (less than 1 m diameter) directional antennas, plus low-gain antennas, are provided as part of the communications system. Communications are relayed through TDRSS or another satellite.[5]

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[5]See Appendix pp. 22-27 of Footnote 1: Hansen, D., Telecommunications Subsystem.

E. COMMAND AND DATA HANDLING

These systems handle radar and other data to be transmitted to Earth, and the commands received by the spacecraft.

F. NAVIGATION

Navigation is via the Global Positioning System (GPS).

SECTION V

SPACE REACTOR POWER SYSTEM CONCEPT

Present concepts of the SRPS are described in Reference 5. Briefly, the SRPS uses as its energy source a fast-spectrum reactor fueled with enriched uranium nitride and cooled with liquid lithium. A shield shadows the rest of the spacecraft from nuclear radiation, and an extendable boom keeps the rest of the spacecraft away from the reactor. The lithium coolant is moved by electromagnetic pumps to a heat exchanger, where it heats one end of a set of thermoelectric elements made of silicon-germanium, doped with gallium phosphide. Waste heat from the colder end of the thermoelectric elements is removed by heat pipes and radiated to space. Electrical power from the thermoelectric elements is conditioned and delivered to the rest of the spacecraft as regulated constant voltage dc. A secondary bus provides power for emergency or special loads, and for use prior to start-up of the main power system.

The reactor operates at constant power and temperature; load changes are accommodated by dumping unneeded power through shunt resistors. This provides load following on a fairly rapid time scale (discussed below). Faster load transients are handled by the capacitors in the radar system.

An aim of SP-100 is to develop a power system that can be scaled from 10's of kWe (kW electric) to 1 MWe. This study, however, was specifically concerned with a 300 kWe system. One proposed concept of a 300 kWe system is shown in Figure 5-1. The various elements lie essentially in a plane. Next to the reactor is the shadow shield. Behind the shield are the flat radiator panels. The primary heat transport system takes the heat from the reactor and conveys it to thermoelectric power conversion modules situated along the radiator panels. The electricity produced is carried by cables along the boom to a control and power conditioning module. Here it is regulated, and unneeded power is dumped. This module also provides the interface to the rest of the spacecraft: commands to the power system and telemetry from the system are transmitted via this interface.

When folded for launch, the SRPS fits in the Shuttle orbiter payload bay or in the shroud of the Titan 4. The proposed version occupies about 9 m of Shuttle bay length (Figure 5-2). The boom is deployed after the system is placed in orbit. The main radiator is deployed as part of the start-up sequence, after the lithium coolant, solid during launch, has been melted. When fully deployed, the length of this candidate SRPS is 25 m; its width, about 20 m (Figure 5-1).

Other candidate SRPS configurations are under consideration (see Reference 5). They differ primarily in the geometry of the radiators and of associated primary heat transport and power conversion equipment. The configuration shown in Figure 5-1 and elswhere in this report is illustrative only.

The boom length and shield thickness are designed to limit the radiation dose delivered to the rest of the spacecraft by the reactor, 1×10^5 rad of gamma radiation and 10^{13} neutrons/cm^2 integrated over the seven year full-power design life of the reactor. (Note: Only five year life is required for this mission.)

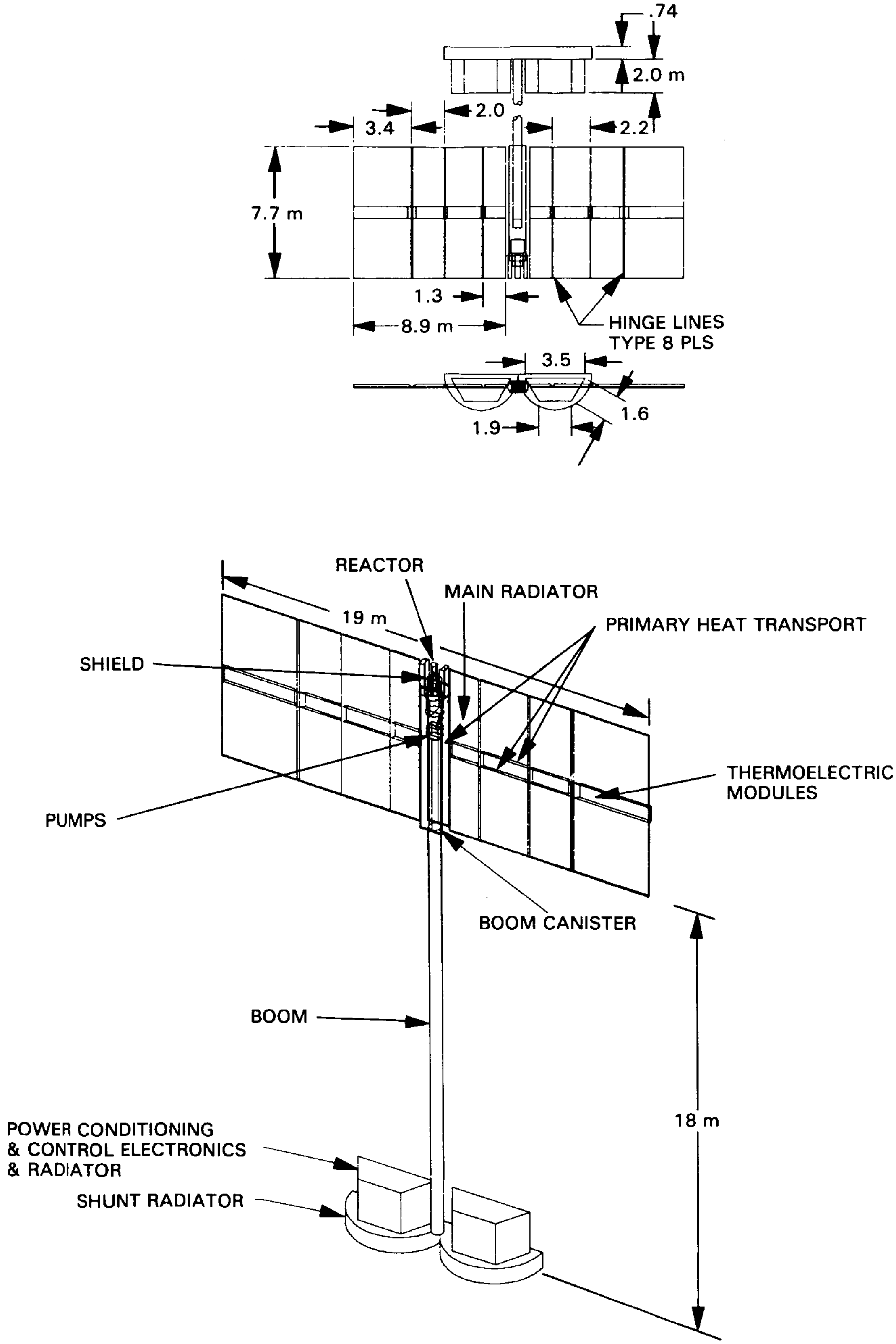


Figure 5-1. Candidate 300 kWe Space Reactor Power System, Operational Configuration

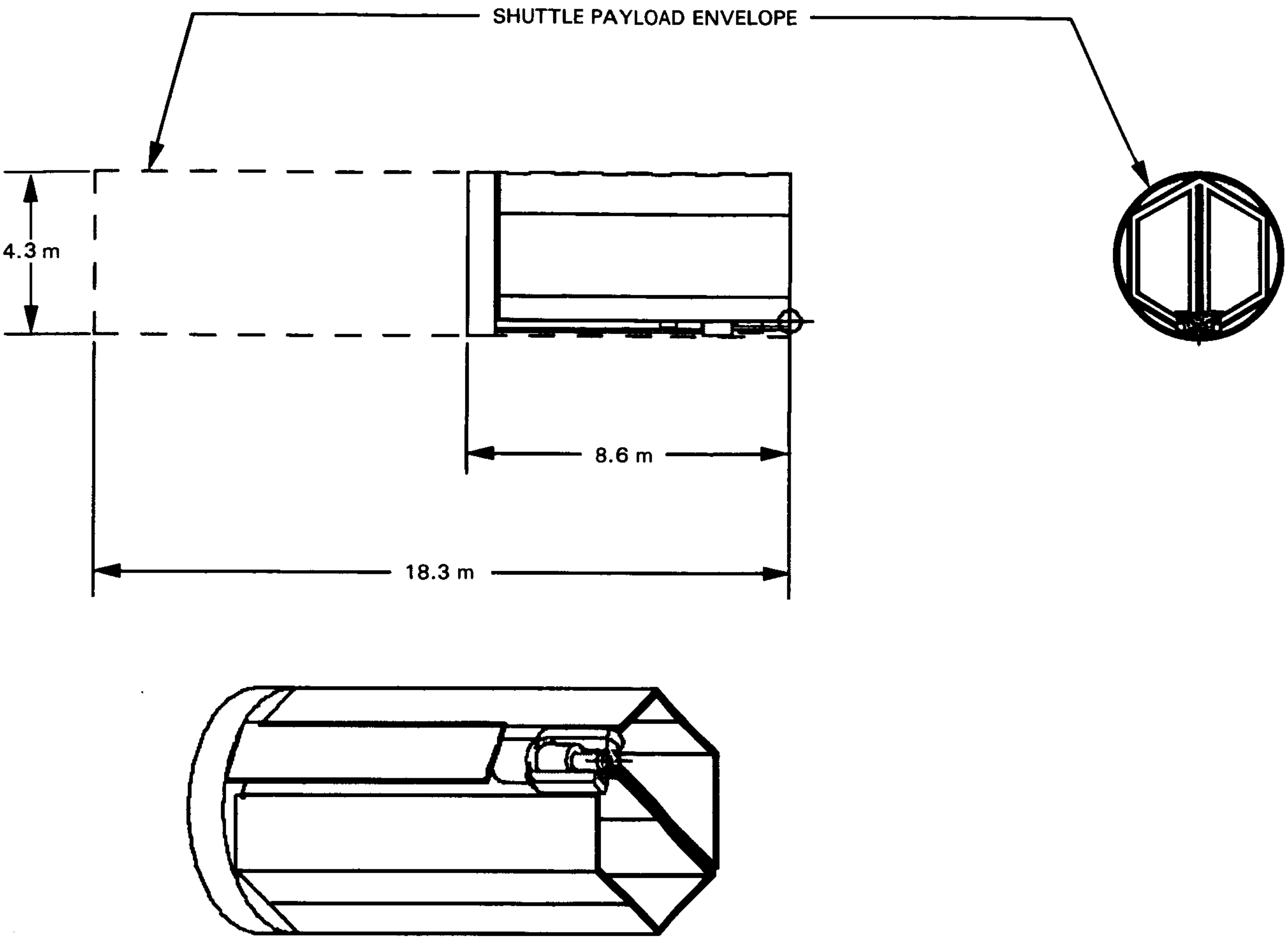


Figure 5-2. Candidate 300 kWe Space Reactor Power System Stowed for Launch

The SRPS main radiator operates at 850-900 K. The thermal radiation delivered to the rest of the spacecraft is not to exceed 1 sun (1.4 kW/m^2).

The SRPS is designed to operate with minimal attention from the ground or the rest of the spacecraft. The start-up sequence, from initiation to full power, takes less than 24 hours. Once up to full power, the SRPS will operate without commands for at least six months; a command then is needed only to inform the SRPS that continued operation is desired.

The SRPS will withstand the natural environment around the Earth for 10 years and still deliver rated power. This includes withstanding ionizing particles at the peak of the Van Allen belts, meteoroids, and debris at all altitudes. It is moderately resistant to hostile threats such as lasers and nuclear explosions and can be hardened further if desired.

Nuclear safety aspects of the SRPS are discussed in Section XIII. Further systems aspects are discussed in Reference 5.

SECTION VI

SPACECRAFT CONFIGURATION

A. OPERATIONAL CONFIGURATION

Two principal operational configurations for the SBR spacecraft were evaluated. Both have as major elements the SRPS, the mission module (containing the communications, command, and attitude control electronics), the radar central power conditioning, the signal processing module, and the radar antenna with its supporting and connecting structure. In one configuration (Figure 6-1a) the SRPS boom axis is vertical; in the other (Figure 6-1b), it is horizontal.

The horizontal boom configuration has advantages in regard to transfer of radiation from the SRPS to the radar antenna and the electronics modules. Since the radar antenna is almost edge-on to the reactor, it provides some self-shielding. Also, the main SRPS radiator partially shields the mission module, signal processing module, and the antenna from reactor radiation. The antenna and the main SRPS radiator are almost edge-on to each other, so heat transfer from radiator to antenna is minimized.

The horizontal boom configuration has, however, difficulties with edge reflection of the radar beam. Much of the spacecraft is outside the radar antenna rectangle. Moreover, to keep these portions of the spacecraft at least 1 m above the lower face of the antenna, they must be offset from the antenna plane. This means that the center of gravity (CG) of the antenna, which is the most massive subsystem, will not lie along the boom axis. The resulting mass assymetry is undesirable from the standpoint of spacecraft dynamics and attitude control.

The edge reflection and mass asymmetry problems were considered important and led to selection of the vertical boom configuration, Figure 6-2, for this study. Appendix B-3 gives additional information.

B. CHEMICAL PROPULSION CONFIGURATION

As discussed below, either chemical or electric propulsion may be used to bring the spacecraft from its Shuttle (or Titan 4) launch vehicle to operational orbit. Chemical propulsion would be used with the spacecraft assembled, but not deployed. It is best located along the spacecraft axis, at the end opposite the reactor (Figure 6-3a).

C. ELECTRIC PROPULSION CONFIGURATION

Electric propulsion would be used with the radar antenna, and the SRPS boom and main radiator deployed. The location of the electric propulsion system was, therefore, examined.

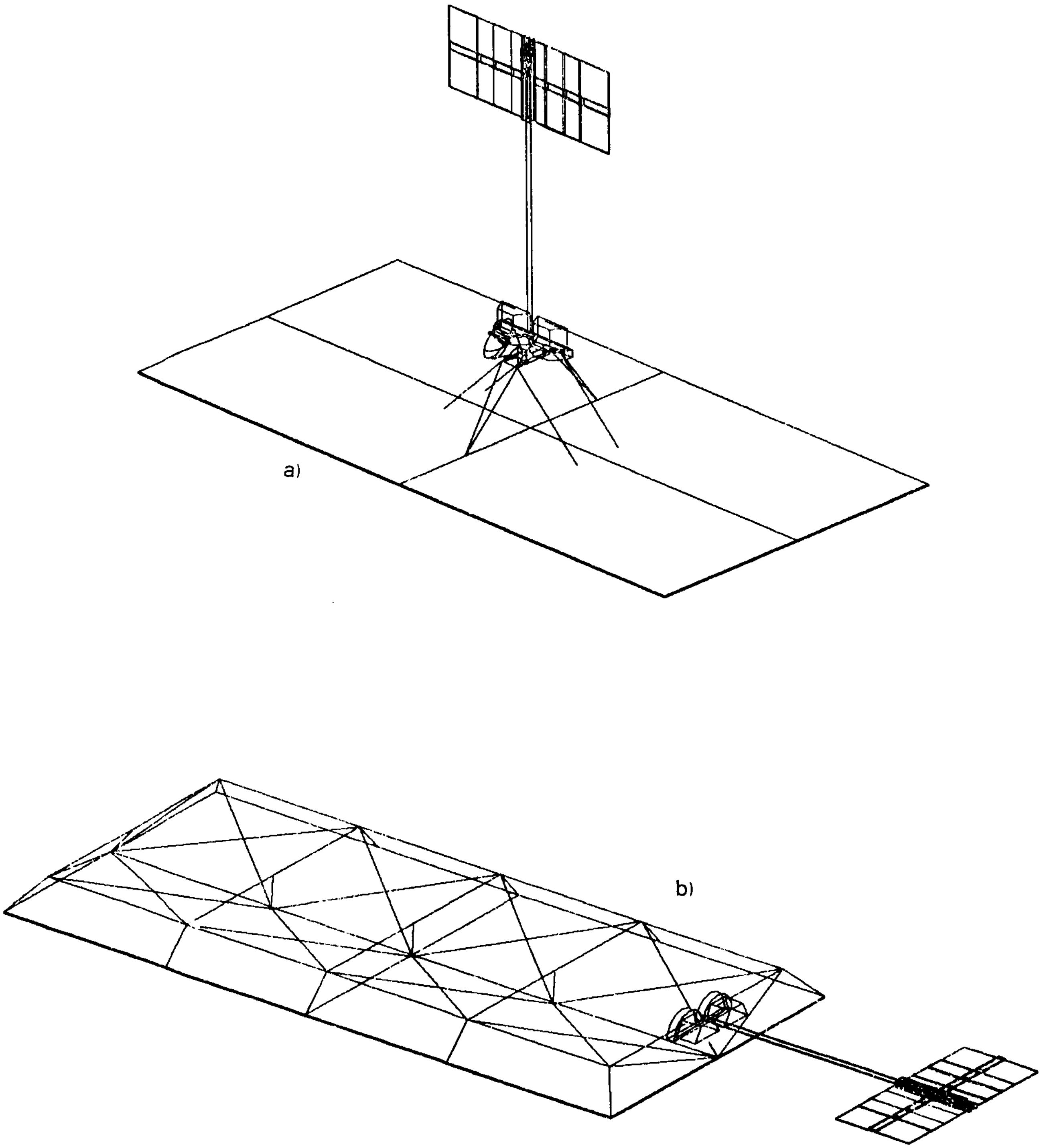


Figure 6-1. Candidate SBR Operational Configurations, (a) Boom Axis Vertical, (b) Boom Axis Horizontal

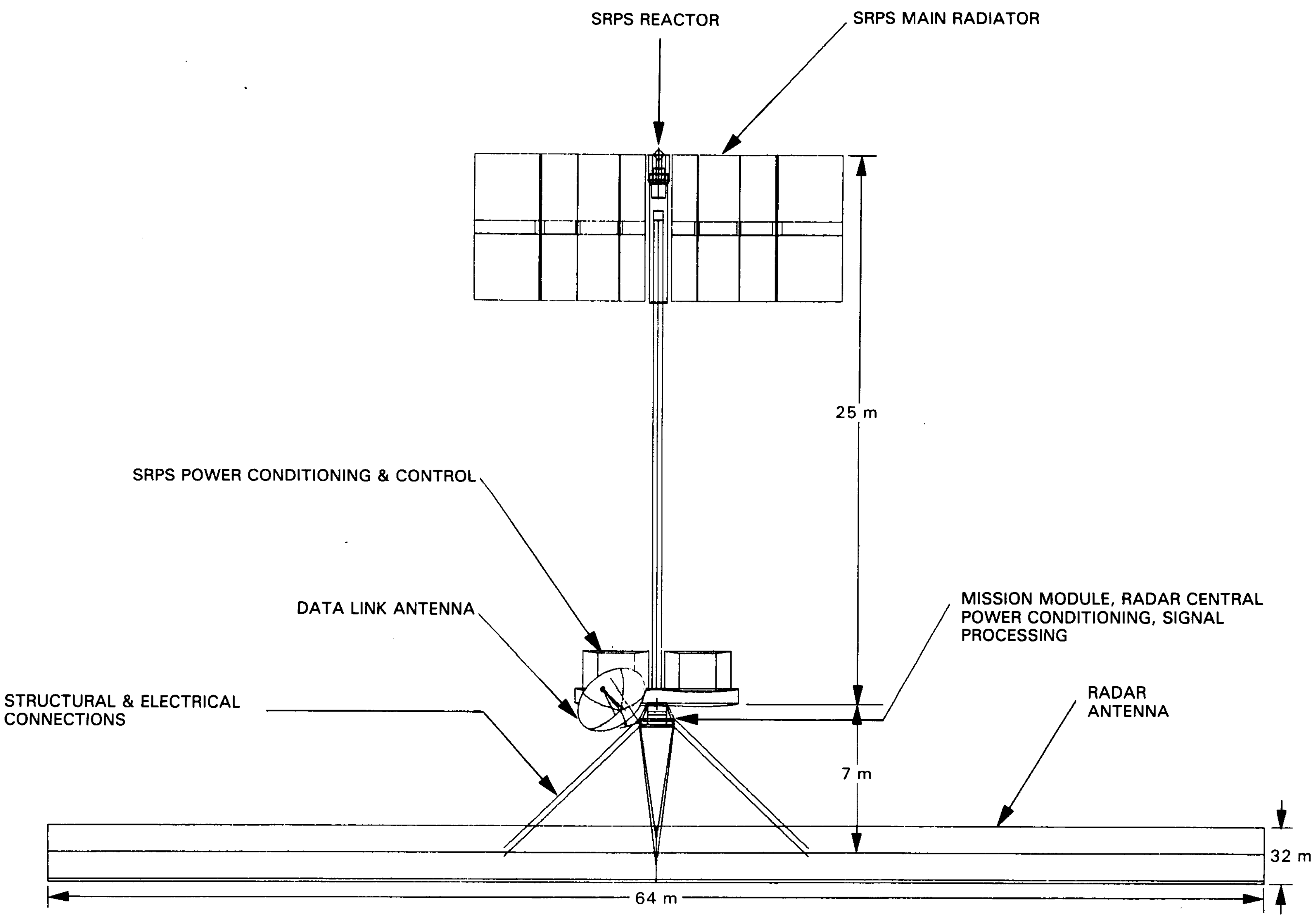


Figure 6-2. Selected SBR Operational Configuration

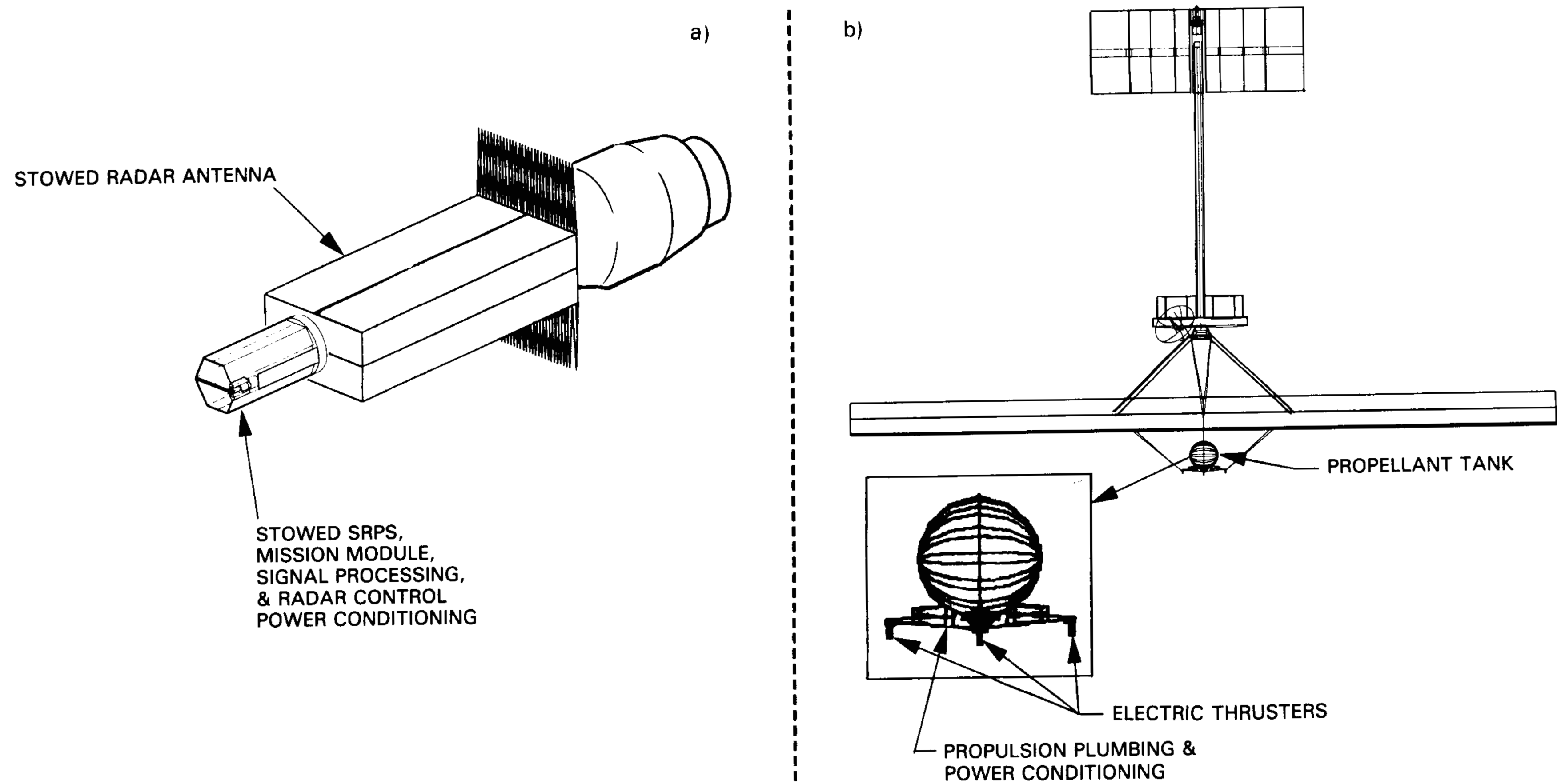


Figure 6-3. SBR Configuration During Propulsion from Assembly Orbit to Operational Orbit, (a) With Chemical Propulsion, (b) With Electrical Propulsion

Four locations were considered. (In the following discussion, the front face of the planar antenna is the face that looks downward during radar peration.) The locations are:

(1) Propulsion just inboard of the four corners of the radar antenna, on the back face. Thrust exhausts away from the antenna back face.

(2) Propulsion just inboard of the four corners of the radar antenna, on the front face. Thrust exhausts away from the antenna front face. To avoid degradation of the antenna pattern, the propulsion modules are jettisoned prior to radar operation.

(3) Propulsion at center of radar antenna, on the front face. Thrust exhausts away from the antenna front face. To avoid degradation of the antenna pattern, the propulsion module is jettisoned prior to radar operation.

(4) Propulsion is located along the boom, close to the spacecraft CG. Thrust exhausts at right angles to the boom.

Advantages and disadvantages of the four locations are:

With locations 1 and 2, a high moment arm is available for attitude control during thrusting. However, if any one of the four propulsion modules fail, the diagonally opposite one must be shut down, and half the thrust is lost. With arcjets, used in the preferred mission profile (below), each module would have only one engine; and failure of one engine would lead to loss of half the thrust, unless a spare engine is provided for each operating engine. If a separate propellant tank is provided for each module, such a loss would also mean loss of half the available propulsive energy. If a central tank is used, plumbing out to the antenna corners will have to be deployed or assembled in orbit.

Location 1 also has the disadvantage that the thruster exhaust is likely to contaminate the SRPS main radiator surfaces and, perhaps, other important spacecraft surfaces. Location 2 avoids this, but requires discarding the propulsion modules before starting radar operation; so they would not be available for any later orbit adjustment, or for unloading the control moment gyros used for attitude control.

Location 3 uses a single propulsion package and, so, avoids the reliability problems just mentioned. It should minimize contamination, because no spacecraft elements are within 90 deg of the exhaust direction. It has the disadvantage that the propulsion module must be jettisoned before the radar is operated.

Location 4, like 3, avoids the reliability problems associated with splitting the propulsion. The exhaust, however, will probably contaminate part of the upper surface of the antenna, which serves as a radiator to dissipate heat generated in the T/R modules and other elements located on the antenna.

Location 3 was selected, and is illustrated in Figure 6-3b. It was felt that propulsion should not be needed after the spacecraft is in operational orbit, and gyros can be unloaded by magnetic torquing of the spacecraft.

D. STOWED CONFIGURATION

Since the spacecraft is to be launched either by the Space Transportation System (STS) or the Titan 4 (see Section VIII), it must fit within the Shuttle Orbiter cargo bay and the Titan 4 shroud during launch, and must fall within the cargo mass limits of these vehicles for the inclination and altitude of the orbit to which it will be launched. The antenna alone, with its associated structure and structural interconnections, has a mass of about 42,000 kg. Multiple launches and assembly in orbit are needed.

For transport by the Shuttle or Titan 4, the spacecraft, excluding propulsion, is divided into five packages. The 32 x 64 m antenna is designed as an assembly of four 16 x 32 m quadrants, placed edge-to-edge. Each quadrant folds into a package 1.8 x 2 x 16 m for stowage. The quadrant, with its support and interconnect structure, is placed in the Shuttle bay or Titan 4 shroud for launch. Each quadrant self-deploys in orbit on command. Figure 6-4 shows one folded quadrant stowed in the Shuttle. Two quadrants with structure will fit within the available volume, but exceed the allowable cargo mass for the orbit needed. The possible alternative of dividing the antenna into thirds has not been examined.

The fifth launch package includes the remaining elements of the SBR spacecraft, less propulsion: the SRPS, radar central power conditioning, the mission module (communications, command, attitude control electronics, etc.), and signal processing. These elements stow within one Shuttle bay (Figure 6-5) or one Titan 4 shroud and require one additional launch.

Because of the Shuttle/Titan 4 cargo mass limitations, the propulsion system cannot be launched with the rest of the spacecraft but must be launched separately and assembled with it in orbit.

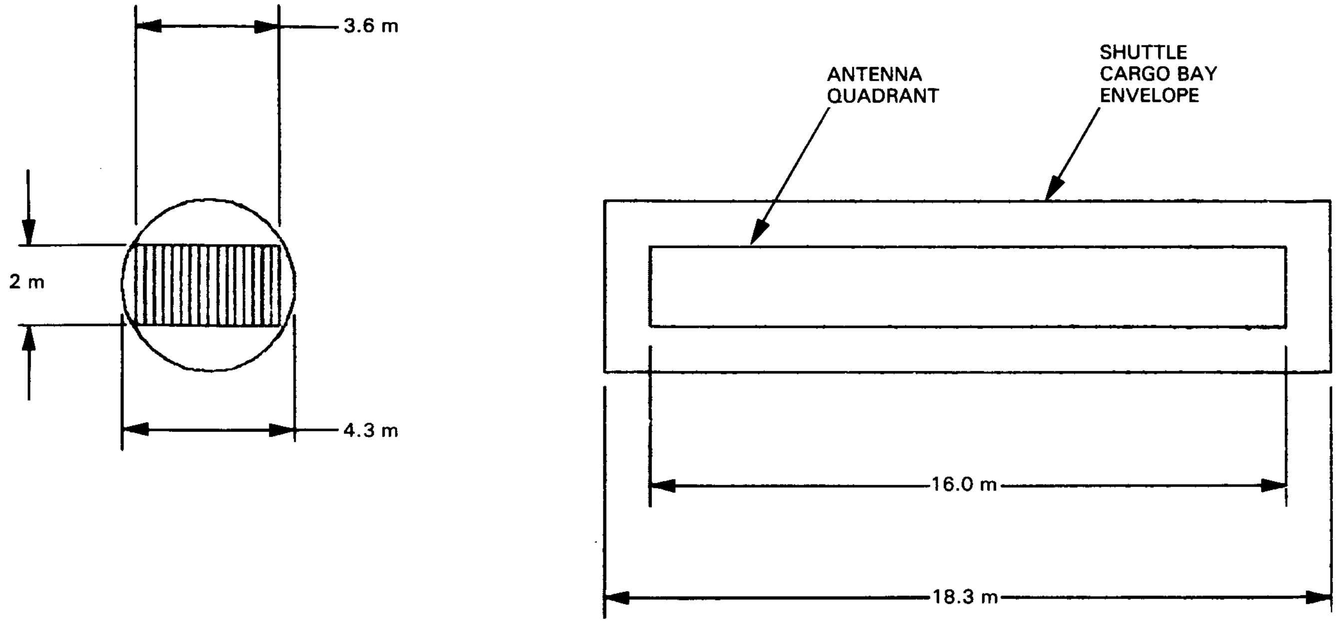


Figure 6-4. Stowed Configuration of SBR Antenna Quadrant in Shuttle Cargo Bay

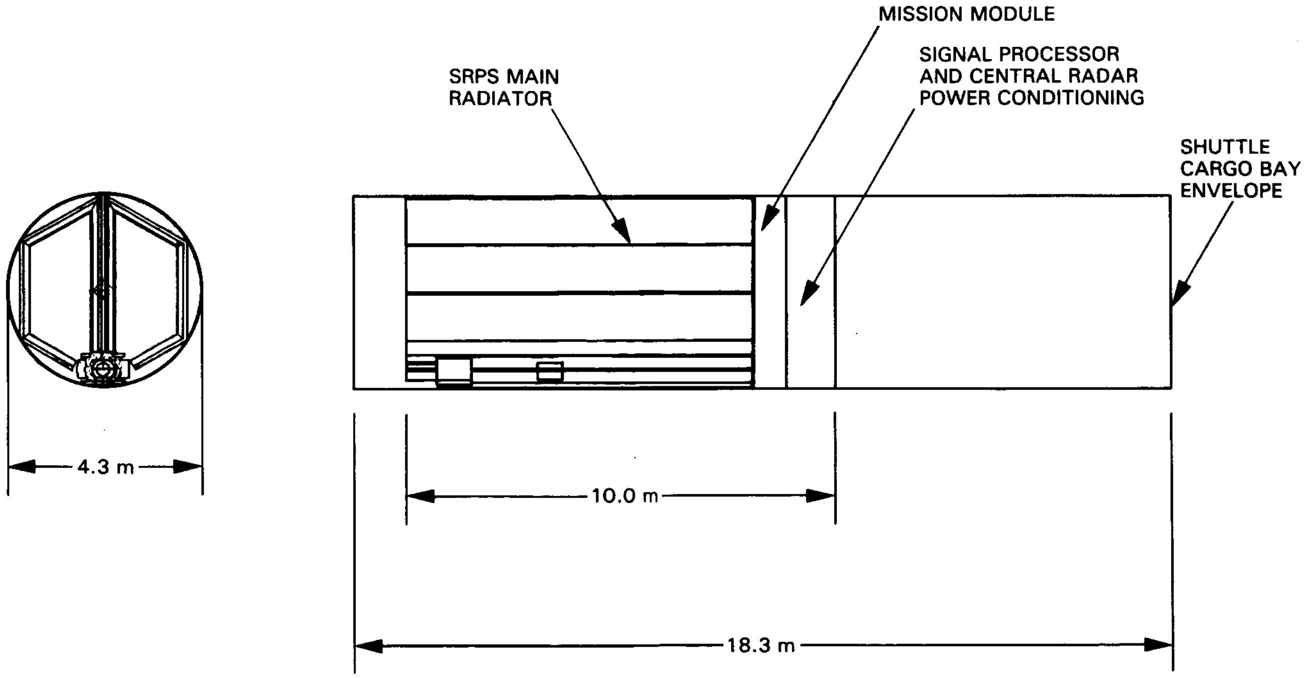


Figure 6-5. SBR Mission Module, Signal Processing Module, and Space Reactor Power System, Stowed in Shuttle Cargo Bay

SECTION VII

MASS BREAKDOWN

Table 7-1 gives a mass breakdown for the spacecraft. The Space Reactor Power System contributes 8,300 kg, the radar system and associated structure 42,800 kg, and remaining items 250 kg, for a total, without propulsion, of 51,300 kg. This is a very large and massive spacecraft!

The SRPS mass is larger than that mentioned in Reference 5; because a large (and therefore heavy) shield is needed to protect the very large antenna from reactor radiation. (See Appendix B-4.)

As already mentioned, either chemical or electrical propulsion may be used to transfer the spacecraft from assembly orbit to operational orbit. If electrical propulsion is used, an electric propulsion module would form part of the spacecraft. Its mass, for the scenario selected (Section IX), is shown in Table 7-1. If chemical propulsion is used, it would be provided by a separate upper stage.

Table 7-1. Spacecraft Mass Breakdown

			kg
SPACE REACTOR POWER SYSTEM			
Reactor			1,650
Shield			1,580
Heat transport			1,450
Power conversion			775
Heat rejection			1,440
System control, power conditioning, and distribution			950
Structure			420
	SRPS total		8,265
MISSION MODULE			
Communications, command, attitude control			250
	Mission module total		250
RADAR			
Radar central power conditioning			250
Signal processing			272
Antenna			
4 antenna quadrants (8818 kg each)		35,272	
Antenna structure		6,300	
Structural interconnects		700	
	Antenna total		42,272
	Radar total		42,794
	SPACECRAFT TOTAL = (without propulsion system)		51,309
ELECTRICAL PROPULSION SYSTEM (if used)			
Electric propulsion unit			4,057
Propellant (NH_3)			5,820
Tank			1,861
	Electric propulsion total		11,738
	SPACECRAFT TOTAL = (with electric propulsion system)		63,047

SECTION VIII

LAUNCH VEHICLE AND PROPULSION

A. LAUNCH VEHICLE

At the beginning of this study, only a launch by the Space Transportation System (STS) was considered. In almost all of the study, STS launch was assumed. Recently, it became apparent, as a result of the Challenger Shuttle accident, that launch by an expendable launch vehicle should be considered; the Titan 4 was chosen because of its close match to STS capabilities. The cargo mass capabilities of the Titan 4 are slightly less than that of the STS; the cargo space available in the Titan 4 matches that of the Shuttle Orbiter cargo bay plus an additional conical volume. The scenario prepared for the Shuttle launch needs little modification for use with the Titan 4. Launch vehicle integration and arrangements for orbital assembly will be significantly different, however.

The dynamic envelope for cargo in the Shuttle Orbiter is 18.3 m long and 4.3 m in diameter. A small space, 1 m long, may have to be reserved at one end for astronaut access to the cargo bay.

Assumed cargo mass capabilities were based on information about the STS available prior to the Challenger accident. At the time of this writing, it appeared that Shuttle capabilities may be significantly reduced because of changes to increase safety; this development was subsequent to most of the work described in this report and is not considered here. Also, the effect of differences between Titan 4 and STS capabilities has not been examined.

The maximum orbital inclination for launch azimuths allowable from Kennedy is 57 deg; the minimum from Vandenberg is 70 deg. Neither reaches the 61 deg selected for the SBR, but the Kennedy launch to 57 deg is closer and would permit higher cargo mass. Launch from Kennedy was, therefore, selected. Two inclinations for the launch vehicle orbit were examined: 57 deg, which minimizes the orbital plane change required, and 28.5 deg, an eastward launch from Kennedy, which maximizes the mass that can be brought to launch vehicle orbit.

The cargo mass the Shuttle can bring to orbit depends not only on the orbit, but also on such variables as the thrust level of the Orbiter main engines, which particular Orbiter is used, whether or not an RMS is carried, the duration of the Shuttle flight, the number of astronauts, use and extent of EVA, various operating reserves, etc. The performance values described in Appendix B-5 were selected for this study.

B. PROPULSION FROM LAUNCH VEHICLE ORBIT TO OPERATIONAL ORBIT

Two propulsion methods were considered for transfer of the spacecraft from the launch vehicle orbit to operational orbit: Chemical and electrical propulsion.

For chemical propulsion, an upper stage using cryogenic hydrogen-oxygen propellant, with an I_{sp} of 444 lbf-s/lbm (4,360 N-s/kg), was assumed. The required propellant mass, derived below, turned out to be 9,600 kg or more, corresponding to a Centaur G-class stage. The NASA decision not to fly Centaur from the Shuttle came when this study was essentially complete, and is not reflected in this report.

For electrical propulsion, the characteristics listed in Table 8-1 were assumed. The propellants and thrusters considered for this mission were ammonia arcjets, with I_{sp} of 1,000 lbf-s/lbm (9,810 N-s/kg), and xenon ion thrusters, with I_{sp} of 3,000 to 4,700 lbf-s/lbm (29,400 to 46,100 N-s/kg). [Performance was also calculated for I_{sp} of 1,100 lbf-s/lbm (10,800 N-s/kg) for arcjets and 2,000 lbf-s/lbm (19,600 N-s/kg) for ion thrusters, but attainment of these I_{sp} values was considered to be a high risk for the time period of interest.]

C. TRAJECTORY AND PROPELLANT MASS

1. Candidate Starting Orbits and Propulsion Modes

Because performance is better with the chemical upper stage or electric propulsion than with the STS or Titan 4, the launch vehicle should not be brought higher than necessary; its cargo capability falls rapidly with increasing altitude. However, the assembly orbit must have an altitude of at least 400 km or so because of orbital decay, as discussed in Section X-C and Appendix B-7. It was assumed that the STS or Titan 4 launch vehicle must bring the spacecraft elements, and the additional propulsion needed, to a circular orbit at this altitude or higher. If spacecraft elements are parked at 400-450 km, some orbital decay will occur before and during assembly; the assembled spacecraft will be slightly lower at the start of the next propulsion burn. A higher initial orbit would be needed if the spacecraft is to be assembled at the Space Station. The Space Station orbit was taken as 500 km circular at 28.5 deg inclination. At this altitude orbital decay during assembly will be small (Appendix B-7).

Three starting orbits were considered for the transfer burn:

(1) 400-450 km altitude, 28.5 deg inclination, circular.

(2) 400-450 km altitude, 57 deg inclination, circular.

(3) 500 km altitude, 28.5 deg inclination, circular (Space Station).

For each starting orbit, three propulsion modes were considered. Each brings the spacecraft to its operational orbit of about 1,100 km altitude circular, 61 deg inclination. (More exactly, 1,088 km, 0.001 eccentricity. The slight difference does not affect the conclusions of this section.) The three propulsion modes are:

(1) Chemical, to the operational orbit.

(2) Electrical, to the operational orbit.

Table 8-1. Electric Propulsion Characteristics Assumed (time period 1995-2000)

ARC JETS

		#	#	#
Propellant	NH_3	NH_3	H_2	H_2
I_{sp}, lbf-s/lbm	1,000	1,100	1,500	1,800
Engine input power, kW	100	100	100	100
Efficiency, PPU	0.96	0.96	0.96	0.96
Efficiency, engine	0.45	0.45	0.54	0.54
Thruster mass, kg	38.8	38.8	38.8	38.8
Engine-associated mass, kg (including thruster)	150	150	150	150
PPU specific mass, kg/kW##	1.4	1.4	1.4	1.4
Tankage & plumbing mass	Per Palaszewski[6]			
Lifetime, h	1,000	1,000	1,000	1,000

ION THRUSTERS

	#				#		
Propellant	Xe	Xe	Xe	Xe	Hg	Hg	Hg
Engine size, cm	50	50	50	50	50	50	50
I_{sp}, lbf-s/lbm	2,220	3,000	3,684	4,710	2,010	3,330	4,260
Engine input power, kW	13	19	29	45	12	29	45 *
Efficiency, PPU	0.92	0.92	0.92	0.92	0.92	0.92	0.92
Efficiency, engine	0.62	0.65	0.75	0.79	0.65	0.77	0.80*
Thruster mass, kg	20.4	20.4	20.4	20.4	20.4	20.4	20.4
Engine-associated mass, kg (including thruster)	80	100	120	170	80	120	170
PPU specific mass, kg/kW##	3.5	2.7	2.2	1.7	3.7	2.3	1.8 *
Tankage & plumbing mass, kg	Per Palaszewski[6]				150 + 0.02 x propellant mass**		
Lifetime, h	5,000	5,000	5,000	5,000	5,000	5,000	5,000

NOTES: Provide redundant engines, enough to cover failure of at least 10% for arc jets and 20% for ion thrusters. Except for a maximum of 1 engine on-axis, engines shall be in sets that balance thrust. Assume that if 1 engine fails its set will be shut down and replaced by a redundant set. This may require increasing the number of redundant engines. Include engine-associated mass for the redundant engines.

\# kW for specific mass are input kW to PPU.

\## Use these columns for parametric studies only.
For low to medium risk:
1,000 lbf-s/lbm is maximum I_{sp} for NH_3 arcjet
1,200 lbf-s/lbm is maximum I_{sp} for H_2 arcjet
3,000 lbf-s/lbm is minimum I_{sp} for Xe ion thruster
2,750 lbf-s/lbm is minimum I_{sp} for Hg ion thruster

6 Palaszewski, B., Hydrogen, Ammonia and Xenon Propellant Feed Systems, JPL IOM 353-PSA-86-098 to Deininger, W., March 11, 1986.

* For values at intermediate I_{sp}, use quadratic interpolation.

** Beatty, Appendix B-6.

(3) Chemical to an intermediate circular orbit at 700 km and the initial inclination, then electrical to the operational orbit.

2. Results

Results are detailed in Appendix B-6 and summarized in Table 8-2. For a 57 deg starting orbit, chemical propulsion is much faster than electric (0.5 day vs. 25 to 72 days). The chemical propulsion is heavier than the electrical, but each can be brought up by a single Shuttle launch. Two-stage propulsion (chemical followed by electric) appears to have no advantage over the simpler one-stage chemical propulsion. The electric propulsion system using ammonia arcjets is about 6,000 kg heavier than those using xenon ion thrusters, but each can be carried in a single Shuttle launch. Transit time is 25 days with the arcjets vs 53-56 days with ion thrusters. Starting from 420 km rather than 450 km altitude has negligible effect on the transit time and propulsion system mass; the same is expected to be true for 400 km.

Note that an initial orbit at 28.5 deg inclination leads to excessive propulsion mass if chemical propulsion is used: 95,000 kg, or at least five Shuttle launches for the propulsion alone. If electric or two-stage (chemical/electric) propulsion is used, the time for transit to the operational orbit is excessive: one to two years. An initial orbit at 57 deg inclination is strongly preferable.

If an initial orbit at 28.5 deg inclination and 500 km altitude is required to permit assembly at the Space Station, ion propulsion appears mandatory to obtain a reasonable propulsion system mass and reasonable number of Shuttle flights to bring up the propulsion system. The very long transit time would have to be accepted. The ion propulsion could be used alone or for the second stage of a two-stage system; the only advantage of two-stage appears to be in nuclear safety, discussed below.

Table 8-2. Time and Propulsion Mass for Transit from Assembly Orbit to Operational*

PROPULSION	CHEMICAL PROPELLANT	CHEMICAL Isp lbf-s/lbm	ELECTRICAL PROPELLANT	ELECTRICAL Isp lbf-s/lbm	TRANSIT TIME, DAYS	PROPELLANT MASS, Mg	PROPULSION SYSTEM MASS, TOTAL, Mg	SHUTTLE SORTIES FOR PROPULSION SYSTEM
Starting Orbit: 450 km 57 deg.**								
Chemical	H_2/O_2	444	----	----	0.5	8.9	12	1.0
Electric, arc	-----	----	NH_3	1,000	25	5.8	12	0.9
Same, 420 km	-----	----	NH_3	1,000	25	5.8	12	0.8
Electric, ion	-----	----	Xe	3,000	53	1.8	6.4	0.5
Same, 420 km	-----	----	Xe	3,000	54	1.8	6.5	0.4
Electric, ion	-----	----	Xe	3,684	56	1.5	5.8	0.5
Same, 420 km	-----	----	Xe	3,684	56	1.5	5.8	0.4
Electric, ion	-----	----	Xe	4,710	71	1.1	5.2	0.4
Same, 420 km	-----	----	Xe	4,710	72	1.2	5.2	0.4
2-Stage	H_2/O_2	444	NH_3	1,000	23	9.4	18	1.5
2-Stage	H_2/O_2	444	Xe	3,000	49	5.5	13	1.1
2-Stage	H_2/O_2	444	Xe	3,684	51	5.2	13	1.0
2-Stage	H_2/O_2	444	Xe	4,710	65	4.8	12	1.0
Starting Orbit: 450 km 28.5 deg.								
Chemical	H_2/O_2	444	----	----	0.5	92	95	5.0
Electric, arc	-----	----	NH_3	1,000	350	80	111	5.9
Electric, ion	-----	----	Xe	3,000	440	15	22	1.1
Electric, ion	-----	----	Xe	3,684	450	12	18	0.9
Electric, ion	-----	----	Xe	4,710	570	9	14	0.8
2-Stage	H_2/O_2	444	NH_3	1,000	340	88	121	6.4
2-Stage	H_2/O_2	444	Xe	3,000	430	20	29	1.5
2-Stage	H_2/O_2	444	Xe	3,684	440	16	25	1.3
2-Stage	H_2/O_2	444	Xe	4,710	550	13	22	1.1
Starting Orbit: 500 km 28.5 deg. (Space Station)								
Chemical	H_2/O_2	444	----	----	0.5	92	95	5.2
Electric, arc	-----	----	NH_3	1,000	330	76	102	5.6
Electric, ion	-----	----	Xe	3,000	440	15	22	1.2
Electric, ion	-----	----	Xe	3,684	450	12	18	1.0
Electric, ion	-----	----	Xe	4,710	560	9.0	14	0.8
2-Stage	H_2/O_2	444	NH_3	1,000	320	82	111	6.1
2-Stage	H_2/O_2	444	Xe	3,000	430	19	29	1.6
2-Stage	H_2/O_2	444	Xe	3,684	440	16	25	1.4
2-Stage	H_2/O_2	444	Xe	4,710	550	13	21	1.2

* Mass transferred, excluding propulsion: 55,000 kg.
Prime power for electric propulsion: 300 kW
Electric propulsion characteristics per Table 8-1
Staging orbit for 2-Stage cases: 925 km circular, same inclination as starting orbit
Operational orbit taken as 1,100 km. For 1,088 km, times and masses will be a few percent less

** Except as noted

SECTION IX

SCENARIOS (MISSION PROFILES)

Three starting orbits and three propulsion modes for transit to operational orbit were considered in the preceding subsection. There were ten basic scenarios for reaching operational orbit, plus variations, such as using Titan 4 instead of STS.

On the basis of the results in Section VIII-C, the following three scenarios were considered best. These are in descending order of preference.

A. PREFERRED SCENARIO: LAUNCH AND ASSEMBLY BY SHUTTLE. CHEMICAL UPPER STAGE TO OPERATIONAL ORBIT

This scenario calls for launch and assembly by the Shuttle and does not involve the Space Station.

Five Shuttle launches are used to bring the spacecraft elements into circular orbit at about 400 km altitude, 57 deg inclination. In each of the first four launches, the Shuttle brings up a quadrant of the SBR antenna with associated structure and leaves it in the orbit mentioned (Figure 9-1). Quadrants are attached to each other as they are brought up. They are allowed to tumble and have passive temperature control. During the fifth launch the Shuttle brings up the rest of the SBR spacecraft (Figure 9-1) and assembles it with the antenna elements. (Assembly methods are discussed in Section X.)

On the sixth launch the Shuttle brings up a Centaur G-class upper stage and attaches it to the rest of the spacecraft. The Shuttle establishes approximate orientation for the Centaur and moves away. At this point the radar antenna is assembled but not deployed; the boom and main radiator of the SRPS are also not deployed (Figure 6-3a).

The Centaur is then activated, acquires proper attitude, burns once to bring the spacecraft to operating altitude of 1,088 km at 61 deg inclination, and burns a second time to circularize the orbit. The Centaur places the spacecraft in the proper orientation, and the radar antenna and boom are commanded to deploy. The Centaur then separates from the spacecraft.

On ground command, the SRPS goes into a start-up mode, which brings it to full operating power, 300 kW electric. The SRPS coolants are thawed; its main radiator panels deployed, both as a part of the start-up sequence. The radar system is checked out and turned on, and the spacecraft becomes operational.

B. ALTERNATIVE SCENARIO: LAUNCH AND ASSEMBLY BY SHUTTLE. ELECTRICAL PROPULSION TO OPERATIONAL ORBIT

Through the first five Shuttle launches (Figure 9-1) and the associated assembly operations, this scenario is the same as scenario A.

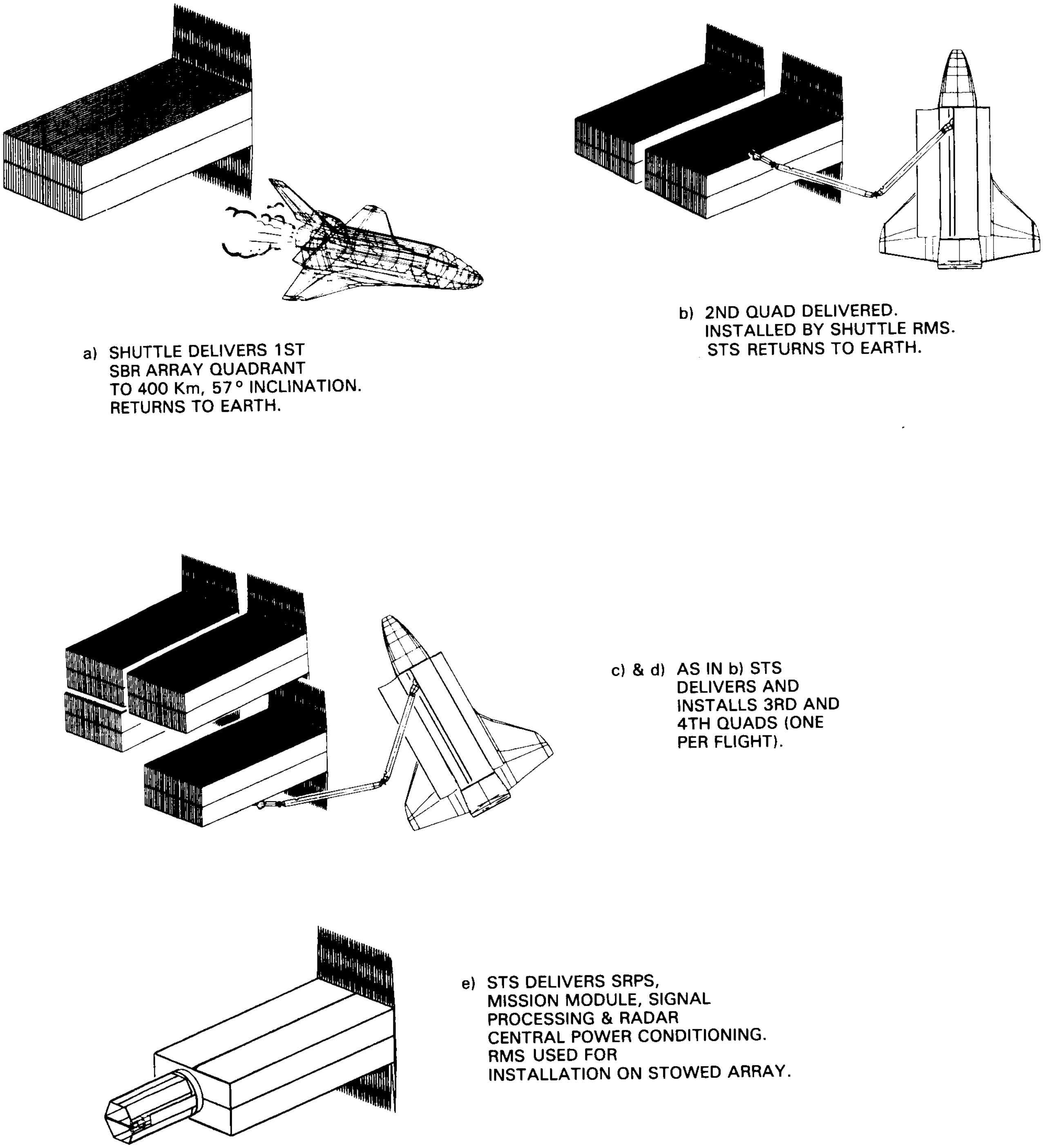


Figure 9-1. Scenario for SBR Launch and Orbital Assembly

On the sixth launch the Shuttle brings up an electric propulsion module, which utilizes arcjet thrusters and ammonia propellant. The propulsion module is attached to the rest of the spacecraft. The radar antenna and boom are commanded to deploy. Figure 9-2 shows the antenna deployment sequence. This deployment and other functions of the spacecraft are checked, then the Shuttle leaves.

The SRPS is started and brought up to power as in scenario A. Coolants are thawed, main radiator panels deployed, both as a part of the start-up sequence. The spacecraft is turned to operating attitude; and the radar is checked out and tested. Then, the spacecraft is rotated to the proper attitude for propulsion; and the electric propulsion system is checked out and started. The propulsion brings the spacecraft to its operational orbit (Figure 6-3b). Propulsion is turned off, and the spacecraft rotated to operating attitude. The radar system is then turned on; the spacecraft becomes operational.

C. ALTERNATIVE SCENARIO: LAUNCH BY SHUTTLE. ASSEMBLY AT SPACE STATION. ELECTRIC PROPULSION TO OPERATIONAL ORBIT

This is the same as scenario B, with the following exceptions:

The propulsion module utilizes ion thrusters and xenon propellant.

Each Shuttle flight brings the spacecraft element to the Space Station, assumed to be in circular orbit at 500 km 28.5 deg inclination, and transfers the element to the Station. Elements are assembled at the Station. The radar antenna and boom are deployed as part of the assembly process.

An Orbital Maneuvering Vehicle (OMV), based at the Space Station, moves the spacecraft to a safe distance (an orbit at about 590 km altitude), using the low thrust mode designed for servicing large observatories. (Maximum acceleration with this payload is about 0.0012 g.) The OMV then moves away from the spacecraft. Start-up is commanded; the rest of the scenario is like scenario B.

D. HYBRID SCENARIO: LAUNCH AND ASSEMBLY BY SHUTTLE, CHEMICAL PLUS ELECTRICAL PROPULSION TO OPERATIONAL ORBIT

Appendix B-8 describes a scenario in which assembly is at 278 km altitude, and two reusable OMVs bring the spacecraft up to 700 km. Then, the SRPS is started; and electric propulsion takes the spacecraft to its operational orbit.

E. SHUTTLE SCENARIO PREFERENCE

These four scenarios were chosen for further consideration because of the propulsion requirements given in Section VIII-C above. Scenario A has the advantage of using a conventional chemical upper stage. It has the disadvantage that the major spacecraft deployment and power-up occur at 1,088 km altitude, so manned intervention to correct any problems would not be possible

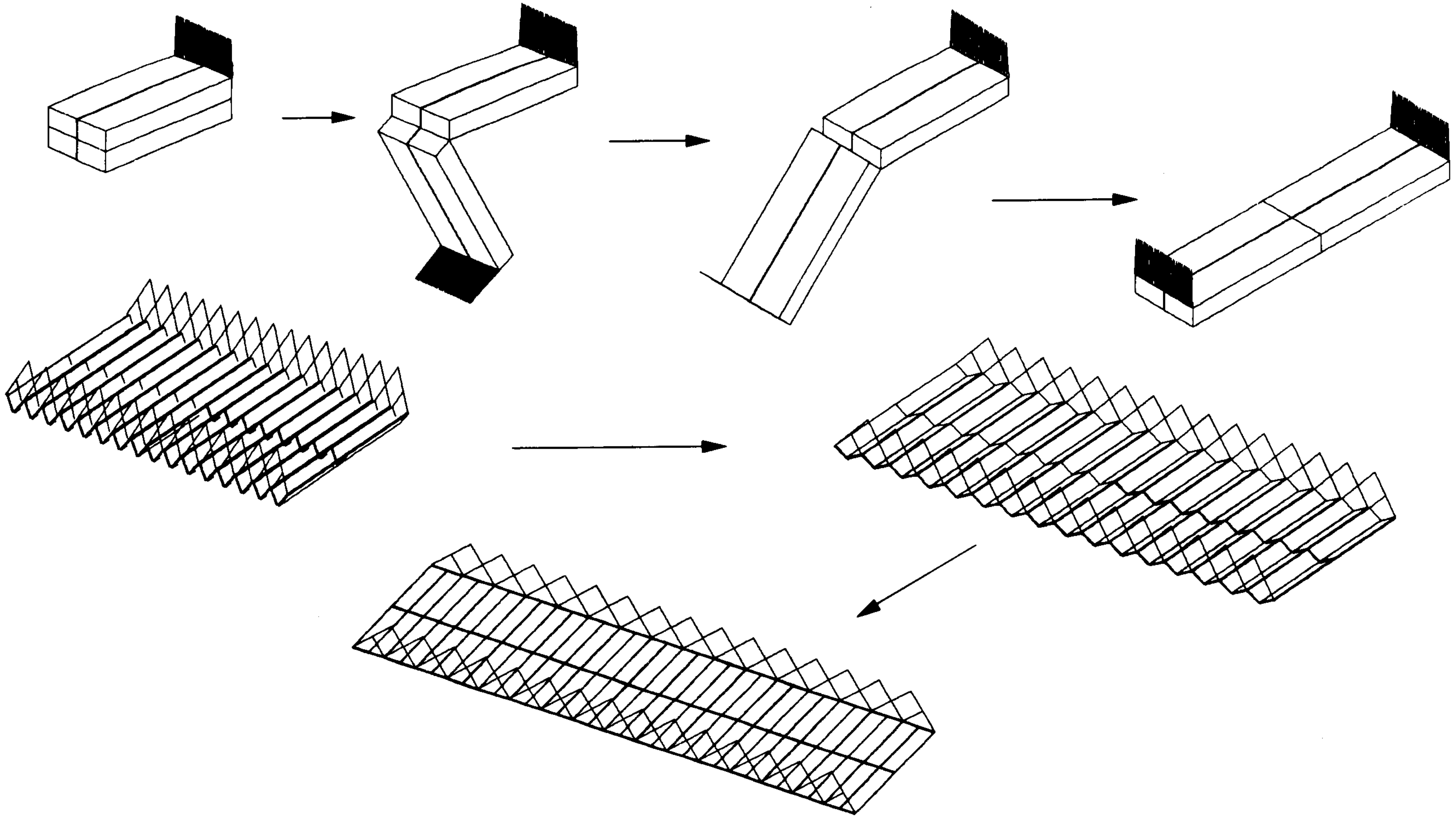


Figure 9-2. SBR Antenna Deployment

with the Shuttle or the Space Station. (NASA recently decided not to fly the Centaur in the Shuttle; this decision came too late to be considered in this study.) Scenarios B and C deploy the spacecraft and start the power system at 400 to 500 km altitude. Checkout will occur here; manned intervention is possible. However, starting the reactor at these altitudes introduces a nuclear safety question, discussed later (Section XIII).

Scenario C is considered less desirable than B, because of the very large orbital velocity increment needed to transfer the spacecraft from the Space Station orbit at 28.5 deg inclination to the radar operating orbit at 61 deg inclination. With chemical propulsion, using cryogenic propellants, six Shuttle launches would be required to bring up the propulsion system, in addition to the five to bring up the spacecraft; and an upper stage (or sequence of upper stages) with much more capability than the Centaur G-prime would have to be developed. With electrical propulsion, ion thrusters are needed, and the transit time to operational orbit is very long (440 days).

Scenario D has the advantage of requiring only three Shuttle flights for spacecraft and propulsion vs the six called for by scenarios A and B. The low assembly orbit of 278 km, chosen to increase mass capability of the Shuttle, leads to an orbital lifetime of a few weeks for the spacecraft elements. It is questionable that three Shuttle launches could be carried out in so short a time. Thus, separate propulsion may have to be added to permit reboost or to place the elements in a slightly higher initial orbit. Also, the Shuttle capability assumed is marginal and exceeds that selected for this study (Appendices B-5 and B-7). If the radar mass could be decreased somewhat, this might be the scenario of choice.

F. TITAN 4 SCENARIOS

Scenarios A and B use the Shuttle for orbital assembly as well as launch. If the Titan 4 is used for launch, one possible assembly approach is to add a Shuttle Flight to provide manned assembly (Section XI-B).

Scenario C is essentially unchanged by the use of a Titan 4 in place of the Shuttle for launch; spacecraft assembly is done at the Space Station.

With Scenario D, Titan 4 could not carry the spacecraft and OMVs to 278 km. Several additional launches would be needed. Assembly of the spacecraft would have to be done by OMVs, or a Shuttle flight added to provide manned assembly (Section IX-B).

SECTION X

SHUTTLE INTEGRATION AND ORBITAL ASSEMBLY

A. SHUTTLE INTEGRATION

Section VI-C describes the elements of the spacecraft that are launched on each Shuttle flight. Figures 6-4 and 6-5 show elements stowed in the cargo bay. Each of these elements conforms to the dimensional limits of the cargo bay, the STS cargo mass limits given in Appendix B-5, the Shuttle CG constraints, and the Shuttle cargo landing mass capability.

Structural support of the SRPS in the Shuttle bay is described in Reference 5. Support of the other spacecraft elements during their individual Shuttle flights has not been addressed.

The SRPS may need a small amount of power while in the Shuttle, and the mission module will almost certainly need some. Both can be accommodated by standard Shuttle cargo electrical provisions. Some telemetry to monitor their state will be desirable; again, standard Shuttle data provisions should do. The other spacecraft electrical elements will be off while in the cargo bay, and probably will not need to be monitored.

If an electric propulsion module is flown, the propellant (ammonia or xenon) will have to be vented or refrigerated while in the Shuttle.

The usual temperature control provided by the Shuttle for cargo should be adequate for all of the spacecraft elements.

If a Centaur is used as an upper stage to place the spacecraft in operational orbit (and if use of Centaur in the Shuttle had not been barred) it would be integrated with the Shuttle as planned for the other Shuttle/ Centaur missions.

B. ORBITAL ASSEMBLY

Scenarios A and B use the Shuttle for both launch and orbital assembly. As described in Sections IX-A and IX-B, the first element brought to orbit is a radar antenna quadrant and attached structure. It is released from the Shuttle, which then moves away. No attitude control or active temperature control is provided.

A Remote Manipulator System (RMS) will be carried on the Shuttle during each subsequent flight. The second, third, and fourth flights each bring up an antenna quadrant, with structure. The Shuttle rendezvouses with the quad or group of quads left by the preceding flights, grasps it with the RMS, and assembles it to the quad still attached to the cargo bay equipment. Astronaut aid at the cargo bay and the RMS arm will probably be needed. The Shuttle releases the newly-assembled group of quads to await the next Shuttle flight.

The fifth launch brings up the SRPS, pre-assembled to the mission module, radar central power conditioning, and signal processing. They are assembled with the antenna quads in the manner described and released to await the next launch.

In scenario A, the sixth flight brings up the Centaur-class upper stage. This is assembled to the spacecraft in the same way as earlier assembly. The Centaur is then placed in desired initial orientation and released. The Shuttle moves away, and the Centaur is turned on by radio command.

In scenario B, the sixth flight brings up the electric propulsion module and assembles it to the rest of the spacecraft, as already described. The Shuttle releases the spacecraft and moves away. Deployment of the SRPS boom and radar antenna, as well as checkout, are commanded by radio. If trouble arises, the Shuttle can return to the spacecraft and attempt to correct it. If spacecraft functions appear normal, the Shuttle leaves; the SRPS, electric propulsion, and other spacecraft elements are then turned on by radio command.

If scenario C is implemented, each Shuttle flight brings a spacecraft or Centaur element to the Space Station and leaves it there. The elements are assembled using Space Station equipment and procedures. An OMV is attached to the spacecraft and takes it to a safe distance away. The OMV returns to the Station and the spacecraft SRPS, electric propulsion, and other systems are started.

If a Titan 4 is used for launch rather than STS, one possibility would be to release the cargo brought up by each Titan 4, and add one or more Shuttle flights at the end to capture and assemble the elements, using the techniques described above.

C. ORBITAL STORAGE

1. Temperature Control

With scenario A or B, the multiple Shuttle launches will take many months. During this time, the spacecraft elements will be in orbit without attitude control or active temperature control. Calculations were performed to determine if their temperature could be held within acceptable limits, considering the possible variations in attitude and in sun/shade cycle (see Appendix B-9). It was assumed that, as a result of tip-off torques, the element spins with a period that is short compared to its orbital period. Results show that, using multilayer insulation having an external layer with proper absorptance/emittance ratio, the temperature can be held within limits of -25 to +25 C. This should be satisfactory for storage of electronics and other components.

The Centaur or electric propulsion module is brought up on the last Shuttle flight, so it need not be stored in orbit.

The SRPS system specification in effect at the start of this study (see Reference 3) did not require that the SRPS be capable of orbital storage. As a result of this work, it was changed to include this requirement (see Reference 4).

2. Orbital Decay

It is desirable to use a relatively low orbit for assembly because the mass capabilities of the STS and Titan 4 fall off rapidly with altitude. However, the spacecraft elements must remain in orbit long enough to complete the assembly. One year was chosen as a conservative estimate for the whole process. The orbital lifetime of each package should therefore be at least 1 year. This might, perhaps, be relaxed to 6 months for the elements brought up in the last few launches.

Table 10-1 and Appendix B-7 show the orbital lifetime calculated for various spacecraft assemblies and attitudes. Aerodynamic drag provides the predominant external torque at assembly altitudes. The spacecraft assemblies are roughly rod-shaped or cylindrical, and will tend to orient with their long axes parallel to the orbital velocity vector, though oscillating around this direction. The drag was taken, conservatively as the average, weighted 2:1, of the drag at 6 deg angle of attack and at 90 deg. The complete spacecraft assembly, undeployed and without propulsion, has the lowest life: One year starting at 400 km, two years starting at 440 km, five years starting at 500 km. An initial altitude of about 400 km appears to be adequate. Some loss in altitude will occur during orbital stays between Shuttle flights, but it should be possible to accommodate this.

Table 10-1. Altitude of Circular Orbit for 1-Year Orbital Lifetime

			Angle of Attack* 6^{o}, spinning		Angle of Attack 90^{o}, spinning		Suggested Values, using weighted mean projected area		
	Mass, kg	Overall shape	Projected Area,** m^2	Required Altitude for 1 year life, km	Projected Area, m^2	Required Altitude for 1 year life, km	Projected Area, m^2	Required Altitude for 1 year life, km	Altitude after 0.5 year, km
1 antenna quad with structure	10,570	Rod	11	370	34	440	19	410	370
2 antenna quads with structure	21,140	Rod	13	350	57	430	28	390	350
Assembled spacecraft, not deployed	51,300	Rod	46	370	157	430	84	400	360

* Angle between orbital velocity vector and plane of maximum projected area.

** Area projected normal to velocity vector

SECTION XI

DYNAMICS AND ATTITUDE CONTROL

The spacecraft requirements, Section III-A, limit the lowest natural structural frequency to 0.01 Hz. The lowest frequency associated with the SRPS alone is of the order of 1 Hz (see Reference 5). It seems likely that lower frequencies will be associated with radar antenna modes. These have not been analyzed.

The attitude control system provides the pointing accuracy of ±0.2 deg needed by the radar. It also provides rotation of ±3.5 deg/orbit about the vertical axis, synchronized with the latitude. Attitude is sensed by an inertial gyro unit. It is controlled by control moment gyros, which are unloaded by interaction of magnetic torquers or current loops with the Earth's magnetic field. (Large amounts of power would be available for magnetic torquing, and large current loops could be provided.) Sun and star or sun and horizon sensors are provided for calibration of the inertial gyro units; and a magnetometer is carried to measure the local magnetic field when unloading is needed. The vertical orientation with the antenna downward is unstable with respect to the gravity gradient; but the control moment gyros provide adequate torque (6,500 N-m) to maintain or restore the desired attitude. (If the SRPS boom is lengthened to 40 meters, the desired orientation will be stable. Trade-offs associated with this change have not been examined. Appendix B-10 provides more detail on attitude control.)

The effect of structural frequencies as low as 0.01 Hz upon the attitude control has been examined only in a cursory way. With simple control techniques, the time required for a 1 radian spacecraft rotation will be about 1,000 sec. With more sophisticated techniques, this can be reduced somewhat. The mission and spacecraft requirements, Sections II and III, do not call for rapid turns.

SECTION XII

THERMAL AND NUCLEAR RADIATION

A. NUCLEAR

The SRPS specification written before this study (see Reference 3) allowed 5 x 10^5 rad of ionizing radiation from the power system to reach the user plane in seven years of reactor operation at full power. The user plane was defined as a circle 4.3 m in diameter, centered on the boom axis, at the interface between the SRPS and the mission module. To shield the radar antenna area, as required in Section II-A, meant that the reactor shield had to be extended radially. Also, Section II-A calls for only 1 x 10^5 rad to reach the antenna from the SRPS. The shield, therefore, had to be thickened. These changes increased the mass of the shield to 3,570 kg, or about 35% of the total SRPS mass of 9,880 kg (Table 7-1).

Partly as a result of the radar antenna requirement, the SRPS system specification was changed to allow only 1 x 10^5 rad of ionizing radiation from the reactor at the user plane (see Reference 4).

B. THERMAL

The selected SBR mission requirements did not limit the thermal radiation delivered to the rest of the spacecraft by the SRPS; the SRPS specification (see References 3 and 4) limited this to 1 sun (1.4 kW/m^2). Analyses (Appendix B-11) showed that the amount delivered depends strongly on the configuration of the SRPS main radiator. The configurations recommended in Reference 5 comply with the specification, when integrated with the rest of the spacecraft as shown in Figure 6-2. With some other SRPS configurations, it may be necessary to insert thermal shielding or lengthen the boom.

The radar antenna, which has the largest projected area and, so, is most difficult to shield from the SRPS radiators, is not at the user plane; it is 7 m further from the radiators. This reduces the thermal radiation from the radiators to the upper surface (back) of the antenna to about 0.6 sun.

It must be recognized, however, that the radiation from the SRPS radiators is only part of the thermal radiation received by the radar antenna. Sunlight can contribute another "sun", and radiation from the Earth can deliver 0.3 sun to the lower surface (face of the antenna). Whether or not the additional heat generated by the T/R modules and other elements on the antenna can be radiated to space, and the elements kept adequately cool, will be an important question for the radar designer. If not, the limit on radiant heat delivered by the SRPS may have to be tightened.

SECTION XIII

NUCLEAR SAFETY CONSIDERATIONS

The SRPS design and the mission profile take nuclear safety as first priority. Except, perhaps, for testing at zero power, the reactor is not operated until a stable orbit is reached. Thus, it will contain insignificant radiological inventory during ground handling, transportation, launch, and ascent to orbit. The reactor is designed to remain subcritical after any credible accident in handling, launch, or ascent, as well as during re-entry, ground impact, and subsequent immersion in water or burial in soil. If a Shuttle or Space Station is used for launch or orbital assembly, the reactor is neither turned on, nor the SRPS operated, in the vicinity of the Shuttle or the Station.

Mission profiles and orbital lifetimes ensure that, even if a failure occurs in flight, the probability of hazardous exposure from the reactor is very low. It is advisable to operate the reactor only at an altitude that allows time in orbit for most of the radioactivity to decay before re-entry. A time of the order of 300 years in orbit provides for substantial decay. After the reactor has operated seven years at full power, and has been shut down for 300 years, the dose rate is calculated at 186 mrem/h at 1 meter from the intact core.

After the work described in this report was completed, the SP-100 Project adopted a policy of placing the SRPS in a permanent storage orbit at end of life. This occurred too late to be reflected in the report.

If electric propulsion is to be used to transfer the spacecraft from its assembly orbit to operational orbit, the safety aspects of starting the reactor at the assembly orbit to provide power for the electric propulsion must also be considered. German and Friedlander (Reference 6) pointed out that as the radioactivity builds up from its initial very low level, so do the altitude and the resulting orbital lifetime; the orbital lifetime may increase fast enough to keep the resulting radioactivity within acceptable limits for an inadvertent re-entry. Figure 13-1 shows the results of calculations of orbital lifetime vs arcjet propulsion time for various initial altitudes (Appendix B-7). Also shown in the figure is the time needed for the resulting radioactivity to decay to the levels mentioned (Reference 7). The orbital lifetime is greater than the time needed for radioactive decay, except during the first few weeks of operation. Whether or not this is acceptable from the safety standpoint remains to be determined. If not, then scenarios B and C, described in Section IX-B, cannot be used.

Once the spacecraft is deployed and stationed at its operational orbit (1,088 km altitude), the calculated orbital lifetime is 425 years if the radar antenna is perpendicular to the orbital velocity vector (the worst case), 775 years if it is at 45 deg, and 4,250 years if it is at 6 deg to the velocity vector (Appendix B-7). The primary external torque on the passive spacecraft in operational orbit is due to the gravity gradient. With the configuration shown in Figure 6-2 (25 m reactor to mission module), the axis of minimum moment of inertia is parallel to the 64 m dimension of the antenna, and the axis of maximum moment of inertia is parallel to the 32 m antenna dimension (Appendix B-10). The spacecraft will tend to orient with the

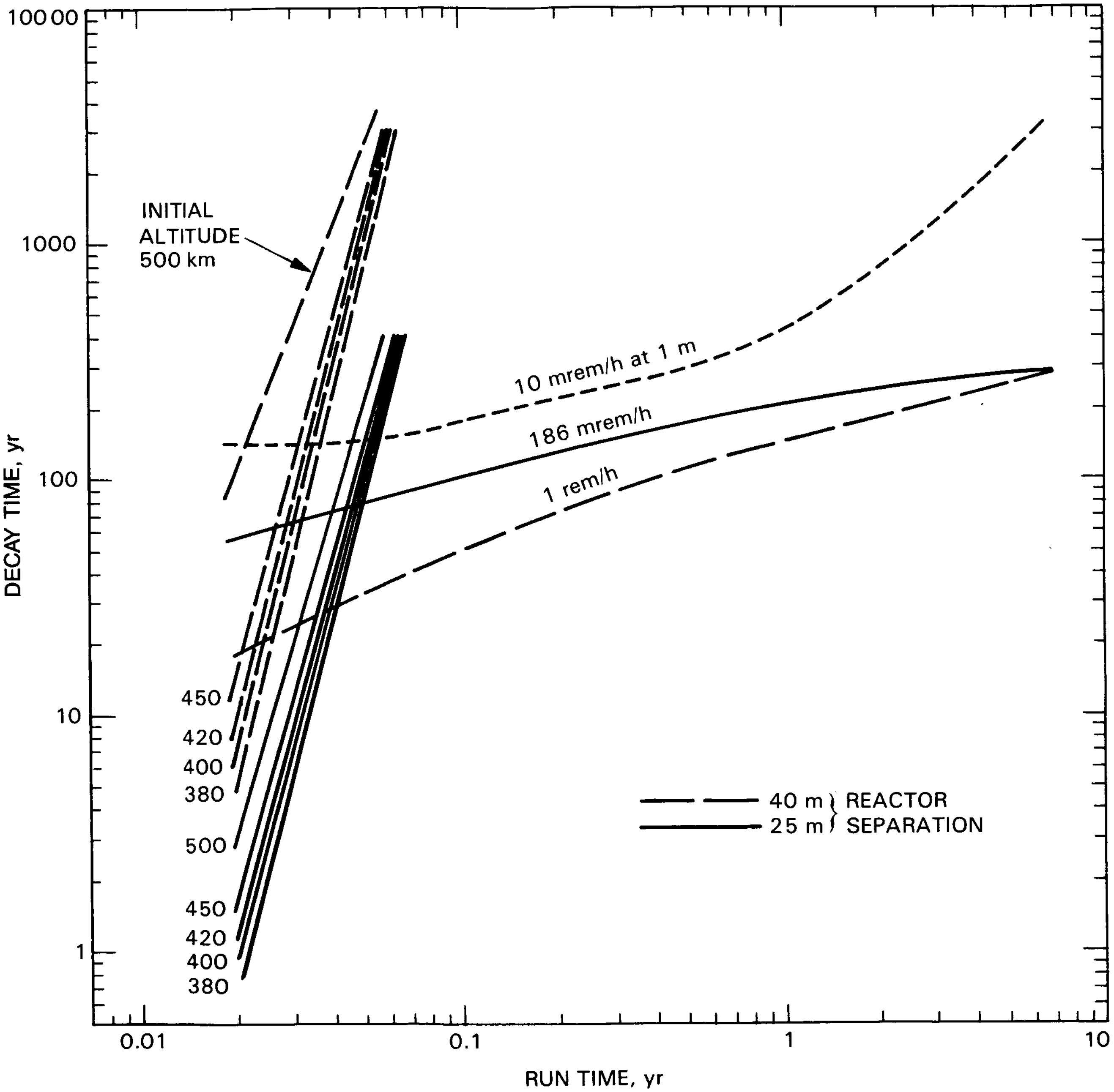


Figure 13-1. Orbital Lifetime After Electrical Propulsion and Time Needed for Decay of Radioactivity, vs Operating Time of SRPS

antenna perpendicular to the orbital velocity, the attitude of maximum drag. Nevertheless, the orbital lifetime will still be 425 years, appreciably greater than the 300 years suggested as desirable after seven years of full power operation.

Appendix B-10 points out that by increasing the separation between reactor and mission module to 40 m, the moments of inertia can be changed so that the stable attitude, due to gravity gradient, will be with the antenna plane parallel to the velocity vector. This would greatly reduce the drag and increase the decay time from the operational orbit. This change would also increase orbital lifetimes near the assembly altitude (Figure 13-1), even though aerodynamic torques are also significant at those altitudes (Appendix B-7). The mass penalty for lengthening the boom and power cable is expected to be relatively small. The increased length would significantly lower frequencies of free vibration. This could probably be avoided by stiffening the boom.

At the end of the spacecraft's mission, the SRPS is turned off by ground command, backed up by an on-board clock. If communication with Earth is lost for more than a preset period, the control system will command shutdown. Two independent shutdown means are provided. If a malfunction occurs during operation, and electric power for control is lost, the safety rods or drums will release and be driven by springs into the "off" position. The reactor is designed to remain intact during re-entry to prevent scattering residual radioactivity.

SECTION XIV

ISSUES AND IMPLICATIONS

This section lists some of the more important findings of the study and the more important issues that have been identified. The findings fall in two categories: those related to design and profile of the mission, and those related to design of the power system.

A. IMPLICATIONS FOR SPACE BASED RADAR MISSION

1. Multiple Shuttle or Titan 4 Launches

Because of the large mass and size of the radar antenna, multiple Shuttle or Titan 4 launches are needed to put the spacecraft into orbit. The problem is exacerbated by the high inclination of the operational orbit, which reduces the cargo mass capability of the launch vehicle. For the scenarios selected in this study, five launches are needed for the spacecraft, plus one for the upper stage or propulsion module.

2. Extensive Orbital Assembly

With multiple launches, extensive orbital assembly is necessary. Techniques for such assembly, presumably based on the Shuttle, will have to be defined and developed if a spacecraft such as the one considered here is to be flown. Tentatively, it is proposed that the Shuttle RMS plus EVA be used.

3. Assembly at Space Station Necessitates Very Large Delta-V (ΔV)

Assembly at Space Station appears undesirable because of the large difference in inclination between the Space Station orbit and the spacecraft operational orbit, which results in a very large orbital velocity increment (ΔV) for transfer between these orbits and correspondingly large propulsion capability. If chemical propulsion is used, at least six Shuttle launches will be needed just to place the propulsion system in assembly orbit. Alternatively, if electric propulsion is used, ion thrusters will be required; and the orbital transfer time will be about 15 months.

4. Extended Orbital Storage

Because multiple launches are required, and assembly at the Space Station is undesirable, each element of the spacecraft must be parked in assembly orbit for an extended time. Spacecraft elements must be designed for orbital storage lasting a year or so. Provision must be included for temperature control and attitude control during this storage or the elements must be designed for storage with only passive temperature control and attitude control. The concept selected in this study employs passive storage.

5. Minimum Altitude for Assembly

A low altitude for orbital assembly is desirable from the launch vehicle standpoint; because launch vehicle cargo mass capability falls off with increasing altitude. However, aerodynamic drag on the spacecraft elements parked in orbit sets a lower limit: Their orbital lifetime must be long enough for all of the required launches and assembly operations. Taking the required time as one year, the required altitude for the proposed spacecraft is about 400 km.

6. Minimum Altitude After Reactor Operation

Once the reactor is turned on, it is considered advisible to turn it off and keep it in orbit long enough for the major portion of the radioactive fission products to decay before re-entry. The decay time suggested by the SP-100 Project is 300 years, after five to seven years of operation at full power. For the radar spacecraft considered, this means that the spacecraft orbit should have a perigee of at least 600-900 km (depending on the stable attitude of the spacecraft in a gravity gradient and the level of radio-activity considered acceptable for re-entry); or it should be reboosted. Since the operational orbit for the proposed mission is 1,088 km, this poses no problem if the reactor is turned on after reaching operational orbit.

If the reactor is to be turned on in assembly orbit, to provide power for electric propulsion, the precautions necessary to ensure safety are not so well defined, and must be examined further.

B. ISSUES FOR DESIGN OF SPACE-BASED RADAR MISSION AND SPACECRAFT

1. Launch Costs

The many Shuttle or Titan 4 launches needed for one SBR spacecraft in the selected scenarios would be costly and would require a large commitment of launch vehicle resources. Accordingly, there will be much incentive to find ways to reduce the antenna mass and to find mission profiles that involve fewer launches.

2. Orbital Assembly Technique and Procedure

Shuttle-based techniques will have to be defined and developed.

3. Technique for Extended Orbital Storage

Temperature control methods are needed for each of the spacecraft elements parked in orbit during the assembly sequence.

4. Choice of Assembly Orbit

Careful attention must be given to the trade-offs affecting the choice of assembly orbit.

5. Safety of Start-up in Low Earth Orbit (LEO)

If start-up of the power system in assembly orbit or other low Earth orbit is contemplated, nuclear safety design, as it pertains to planned or unplanned re-entry, will be very important.

6. Dynamics and Attitude Control

The very large, flat, radar antenna, in conjunction with other flexible elements such as the SRPS boom and main radiator panels, may lead to structural vibration modes with very low frequencies. Interaction of these structural elements (especially the antenna) with attitude control will need to be investigated.

7. Maneuverability

Because of the low structural frequencies expected, rapid changes in spacecraft attitude will probably not be possible. Also, it may be difficult to provide adequate antenna strength to withstand moderate accelerations. The maneuverability of the spacecraft is likely to be limited.

8. Thermal Radiation from SRPS to Rest of Spacecraft

It is not clear whether electronic components on the radar antenna can be kept within permissible temperature limits when the antenna is receiving the currently allowable one sun from the SRPS, plus thermal radiation from the sun and Earth. This will have to be investigated. It is possible that the limit of 1 sun from the SRPS may have to be lowered.

9. Allocation of Dosage of Ionizing Radiation

The stated mission requirements (Section II-A) allocate to the SRPS only 1% of the total dosage of ionizing radiation delivered to the antenna, reserving 99% for natural and hostile sources. Further thought should be given to this allocation. (However, the stated allocation does not place a great burden on the SRPS design.)

10. Survivability

If there are requirements for the spacecraft to withstand hostile action, the ability of the radar antenna and other spacecraft systems to withstand these threats will need to be examined.

11. Radar Power Level, Size, and Mass

The design of the spacecraft is driven by the power level of the radar and the size of the radar antenna. Those chosen in this study led to a very large and very heavy spacecraft, requiring multiple Shuttle launches, which is unlikely to be the first operational space-based radar spacecraft. When designing the radar system, the implications for the number of launches and the extent of orbital assembly need to be considered carefully.

C. IMPLICATIONS FOR SP-100

1. Extended Orbital Storage of Power System

Like the rest of the spacecraft, the Space Reactor Power System will have to be stored in assembly orbit for six months to a year before it is turned on. This capability was not required in the SRPS system specification in effect before this study; the specification has been changed because of this finding.

Thermal analysis shows that even without attitude control, the temperature of critical SRPS components can readily be kept within acceptable temperature limits by using multilayer insulation and an exterior surface with appropriate absorptance/emittance ratio.

2. Mission-Specific Shield Configuration

The constraint on dosage of ionizing and neutron radiation from the reactor to other parts of the spacecraft is stated in the system specification in terms of specific values to a specific area. (Before this study: 5×10^5 rad and 1×10^{13} neutrons/cm^2, integrated over seven years operation at full power, at any point on a circle 4.3 m in diameter, centered on the boom axis at the interface between the SRPS and the mission module.) Requirements for the SBR mission make it clear that both the allowable dosage and the area where the allowance applies will vary from mission to mission.

3. Thermal Radiation from SRPS to Rest of Spacecraft

The system specification limits the thermal radiation from the SRPS to the rest of the spacecraft to 1 sun (1.4 kW/m^2). Analysis indicates that this requirement can be met with some SRPS radiator configurations; it may be difficult to meet it with others. The requirement may, therefore, be important in the selection of the radiator configuration.

4. Reactor Throttle-Down Not Necessary

In the design of the power system, the ability to operate with the reactor power throttled down to low levels may not be easy to provide. For the mission examined here, this capability does not appear necessary.

SECTION 15

REFERENCES

1. Jaffe, L., et al., Nuclear Power for a Reusable Orbital Transfer Vehicle, JPL Publication 86-51, Jet Propulsion Laboratory, Pasadena, California (in publication).

2. SP-100 Air Traffic Control Space Based Radar (SBR) Mission Application Study: Final Report, JPL Contract 956969, General Electric Co., Space Systems Division, Valley Forge, Pennsylvania, October 31, 1984.

3. Technical Specification for the SP-100 Space Reactor Power System (SRPS): Exhibit 1, DOE San Francisco RFP No. DE-RP03-86SF16006, October 22, 1985. (Note: This is not the latest issue of the specification, but, rather, the issue in effect during most of the study described in this report.)

4. Technial Specification for the SP-100 Space Reactor Power System (SRPS): Exhibit 1, Revision 4, DOE San Francisco RFP No. DE-RP03-86SF16006, October 20, 1986.

5. Jaffe, L., et al., System Aspects of a Space Nuclear Reactor Power System, JPL Publication 87-12, Jet Propulsion Laboratory, Pasadena, California (in publication).

6. German, D., and Friedlander, A., Nuclear Safe Orbit Raising Analysis, JPL Contract 956817, Report No. SAIC-85/1911, Science Applications International Co., Schaumberg, Illinois, November 1985.

7. Boudreau, J., Safety of Reactor Start-Up in (High) Shuttle Orbit, Los Alamos National Laboratory Document LA-CP-86-262(SPTN 123), Los Alamos National Laboratory, Los Alamos, New Mexico, 1986.

APPENDIX A

SURVIVABILITY AND RELATED MATTERS

(Appendix A is classified,
and is distributed separately.)

Since the completion of this report, Appendix A, because of its classified status, has been recreated as a separate document printed by the Jet Propulsion Laboratory. The title is: Survivability Aspects of a Space Reactor Power System; the document number is JPL D-3758.

This document can be made available to individuals who hold a secret clearance and have a need for the information by contacting the SP-100 Project Office, Jet Propulsion Laboratory, Pasadena, California.

APPENDIX B

DETAILED EXAMINATION AND ANALYSIS

This appendix is an assembly of documents recording some of the detailed work which backs up the body of the report.

The documents are individually dated and show concepts and analyses as they evolved during the study. There are, therefore, some differences in ideas and results among the appended documents, as well as between them and the body of the report prepared later. One example is the power system configuration. A number of configurations for that system were considered; illustrations and calculations in this appendix represent several of these configurations.

In a few cases, figures and other portions of the original documents represented in this appendix have been transferred to the body of the report or omitted as noted.

B-1. RADAR REQUIREMENTS AND CHARACTERISTICS

T. Fujita

November 1983 to January 1986

This section consists of 5 conference reports containing inputs from Lincoln Laboratory, Massachusetts Institute of Technology. Short portions of the January 9 and January 21 reports have been omitted.

CONFERENCE REPORT

JET PROPULSION LABORATORY

REPORT NO. TES-354-85-114

SUBJECT Spaced-Based Radar (SBR)--MIT Lincoln Laboratory

PROJECT ______ CONTRACTOR ______ CONTRACT or

ACTION REQUIRED BY ______

TELECON Initiated by T. Fujita CONFERENCE at ______ Date of Occurrence 27 November 1985	Report Prepared by T. Fujita TF Date Prepared 29 November 1985
Participants	**Distribution**
Dr. Gerry Tsandoulas MIT Lincoln Labs Tosh Fujita, JPL	R. Beatty, R. Caputo, E. Chow, R. Ewell, R. Ferber, L. Jaffe, R. Manvi, J. Mondt, B. Nesmith, J. Roschke, J. Rose, J. Stevens, J. Stallkamp, G. Stapfer, V. Truscello

BACKGROUND

Information regarding mass and volume envelopes for selected SP-100 candidate missions is being gathered as part of a study involving SP-100 configurational packaging approaches. The SBR concept proposed by MIT Lincoln Laboratory was used in an earlier JPL mission study (see briefing package entitled "SP-100 SBR Study, Final Review," by Ross M. Jones, dated July 16, 1985).

The purpose of the call documented in the present report was to obtain inputs regarding larger systems corresponding to the 300 kW_e baseline selected for Phase II as well as any updates regarding further work on the Lincoln Lab (LL) SBR concept.

The inputs provided by Dr. Gerry Tsandoulas of LL consisted of (1) results of a study at LL for a 120 kW_e SBR and (2) projections for a larger 300 kW_e SBR system.

LL SBR STUDY FOR A 120 kW_e SBR

The LL study was based on a 120 kW_e (prime power) L-band SBR that was considered to be approximately the largest size system that could be launched with a single shuttle flight. The mission was predicated on a shuttle capacity of 53,700 lbs (24,410 kg) corresponding to 109% engines and a 57° orbit. Power was provided by GaAs solar cells (17% efficiency) with energy storage in nickel-hydrogen batteries. Electric propulsion is used to transport the system from the shuttle altitude to the 600 n.m. SRB orbit. The dimensions of the phased array are 16 m x 32 m. (Note that this array is twice the size of the 8 m x 32 m array treated in the earlier JPL study).

A mass breakdown from the LL study is given below:

	lbs	kg
Phased array	19,400	8,818
Power (GaAs/$Ni-H_2$)	11,600	5,272
Orbit Raising (Electric Propulsion)	4,700	2,136
Structures	3,600	1,636
Signal Processing/Data Link	1,100	500
IR Sensor	2,000	909
Miscellaneous plus 10% contingency	5,800	2,636
TOTAL:	48,200	21,907

The system requires 28 volts d.c. to be supplied to microwave generators located within the phased array. If the GaAs/$Ni-H_2$ power subsystem were to be replaced with a nuclear SP-100 of ≈3000 kg, a mass savings of ≈2000 kg would result. Thus, it would appear that a slightly larger system might be accommodated with the nuclear system. A detailed study would be required to determine the extent of the SBR size increase.

The unit mass of the phased array is

$$\frac{\text{ARRAY MASS}}{\text{ARRAY AREA}} = \frac{8818 \text{ kg}}{16 \text{ m} \times 32 \text{ m}} = \frac{8818}{512} = 17.2 \text{ kg/m}^2$$

It is noted that the earlier JPL study was based on an array size of 256 m^2 (8 m x 32 m) and an array mass of 6300 kg. This provides a higher unit mass of 24.6 kg/m^2. Dr. Tsandoulas was unable to explain the difference, but thought that the earlier JPL study may have been based on a heavier microwave generating setup, which LL had formerly used.

PROJECTIONS TO A ~300 kW_e SBR SYSTEM

For a larger power system, Dr. Tsandoulas believes that a system with a ~ 10 db increase in sensitivity would be of interest. This could be achieved with a four-fold increase in area and a factor of 2.5 increase in power. This system, as visualized by Dr. Tsandoulas, would have a size of 32 m x 64 m and a power level of ≈300 kW_e (2.5 x 120 kW_e).

According to Dr. Tsandoulas, LL considers a length of 32 m to be about the upper limit for conventional rigid structure approaches. The driver is the flatness requirement for the array. Beyond 32 m, advanced approaches (e.g., "dynamic membrane") would have to be invoked. These approaches, if successful, could result in a guesstimated 20-40% mass savings as compared to conventional rigid-body approaches. However, these advanced structural concepts are unproven and will require an intensive development effort.

Based on the unit mass of the 120 kW_e array, a mass of 35,272 kg (8818 kg x 4) is estimated for the 300 kW_e array. If advanced structures can provide a savings of 40%, the mass would be 21,163 kg. This weight for the 300 kW_e phased array is approximately the same as the total mass for the 120 kW_e system. Thus, based on mass assumptions used by LL, the 300 kW_e SBR concept visualized by Dr. TRsandoulas will require at least two shuttle flights and some on-orbit assembly.

TF:mr

CONFERENCE REPORT

JET PROPULSION LABORATORY

REPORT NO. TES-354-85-132

SUBJECT Lincoln Labs Spaced-Based Radar (SBR) Packaging Volume

PROJECT ______ CONTRACTOR ______ CONTRACT or

ACTION REQUIRED BY ______

TELECON Initiated by T. Fujita	Report Prepared by T. Fujita TF.
CONFERENCE at	Date Prepared 31 December 1985
Date of Occurrence 30 December 1985	
Participants	**Distribution**
Darryl Weidler, MIT Lincoln Labs Tosh Fujita, JPL	R. Beatty R. Caputo E. Chow W. Deininger R. Ewell R. Ferber L. Jaffe' R. Manvi J. Mondt B. Nesmith J. Roschke J. Rose J. Spanos J. Stevens J. Stallkamp G. Stapfer V. Truscello

BACKGROUND

As a follow-up to a previous telecon (see Conference Report No. TES-354-85-127 dated 12 December 1985), a call was placed to obtain additional details regarding the packaging volume of the 16 m x 32 m SBR antenna. The viewgraphs showing the antenna packaging arrangement (as mentioned in the previous report) have been requested from Graeme Aston, who had previously received this material. Since this material has not yet been located, further details as reported herein were sought.

PHASED ARRAY PACKAGING ARRANGEMENT

In checking through his notes, Weidler found that the height, width, and length of the stowed array was [1.78 m (70") x 2.39 m (94") x 16 m]. the array was composed of 18 panels having dimensions of 1.78 m x 16 m, where 18 x 1.78 m = 32 m. Weidler had previously estimated that there were 16 panels having dimensions of 2 m x 16 m (see Report No. TES-354-85-127), but the later values as given herein reflect a more careful review of his notes. In the stowed configuration, the width permitted for each panel is 2.39 m/18 panels = 0.133 m. Each panel and associated folding truss structure must fit within this 0.133 m (5.22") allocation.

It is noted that two of the folded 16 m x 32 m arrays can be fitted into the shuttle bay, e.g., stacking two rectangular packages results in an outer envelope of 3.56 m x 2.39 m. The maximum length as given by the diagonal of this rectangle is $\sqrt{(3.56)^2 + (2.39)^2} = 4.3$ m, which corresponds to the diameter of the useable cargo bay. The mass of the 16 m x 32 m array is 19,400 lbs (8818 kg). Therefore, two arrays would have a mass of 38,800 lbs (17,636 kg).

As a baseline for the 32 m x 64 m SRB system corresponding to the 300 kW_e SP-100 it was suggested that four of the 16 m x 32 m arrays be linked together. This would require interconnecting structures and probably an electronic compensation system (see previous report). Weidler agreed that this would constitute a reasonable starting point. For this baseline, two shuttle flights would be required for the phased array and a third flight would be required for the SP-100 and remainder of the system.

CONFERENCE REPORT

JET PROPULSION LABORATORY

REPORT NO. TES-354-86-005

SUBJECT Scaling of Lincoln Laboratory Spaced-Based Radar Data

PROJECT ______ CONTRACTOR ______ CONTRACT or

ACTION REQUIRED BY ______

TELECON Initiated by T. Fujita CONFERENCE at ______ Date of Occurrence 9 January 1986	Report Prepared by T. Fujita Date Prepared 9 January 1986
Participants	**Distribution**
Darryl Weidler) MIT Gerry Tsandoulas) Lincoln Labs Tosh Fujita, JPL	R. Beatty, R. Caputo, E. Chow, W. Deininger, R. Ewell, R. Ferber, L. Jaffe, R. Manvi, J. Mondt, B. Nesmith, J. Roschke, J. Rose, J. Spanos, J. Stallkamp, G. Stapfer, J. Stevens, V. Truscello

BACKGROUND

An approach to estimating the mass of the 300 kW_e Spaced-Based Radar (SBR) powered by the SP-100 is to scale data provided by Lincoln Laboratory (LL). Discussions with Jim Stevens, who is developing mass estimates, indicated the need to clarify some of data previously gleaned from LL (e.g., Conference Report TES-354-85-114, dated 29 November 1985). The purpose of this telephone conference was to obtain scaling information and clarify associated configurational details.

ARRANGEMENT OF RADAR PANELS

The 32 m x 64 m radar for baseline preliminary analysis purposes is taken to be composed of four 16 m x 32 m panels, where the design of the panel is based on an existing LL arrangement. There are two alternate arrangements for the panels. In one arrangement, the four panels could be placed side-by-side with the 64 m side composed of four 16 m segments. In the other arrangement, the four panels could be located so that each panel forms a quadrant of the 32 m x 64 m antenna. Both Weidler and Tsandoulas felt that the quadrant arrangemment would probably be advantageous from a power distribution standpoint. Therefore, it is suggested that the quadrant arrangement be chosen for the baseline.

SCALING OF SIGNAL PROCESSOR/DATALINK

For the 16 m x 32 m SBR, this category was given a mass of 1100 lbs (500 kg). In searching through his notes, Weidler found that he had used the same value for the smaller 8 m x 16 m SBR, which implies that the mass remains essentially constant with power level and antenna size.

Weidler's notes indicated that the contents of this category were

	lbs	kg
Signal Processing	600	273
Attitude Control	100	45
Telemetry and Command	200	91
Thermal Control	200	91
	1100	500

Tsandoulas noted that there would probably be a very small increase in mass with size, but that assuming this mass to be 1100 lbs (500 kg) for the 32 m x 64 m system is a reasonable starting point. Although the larger array has many more modules, Tsandoulas noted that there would only be small changes in signal processor mass. Regarding attitude control, the phased array evidently includes its own attitude control.

CONFERENCE REPORT

JET PROPULSION LABORATORY

REPORT NO. TES-354-86-013

Page 1 of 2

SUBJECT Criteria for Shaping Antennas for Space-Based Radars

PROJECT ______ CONTRACTOR ______ CONTRACT or

ACTION REQUIRED BY ______

TELECON Initiated by T. Fujita CONFERENCE at ______ Date of Occurrence 21 January 1986	Report Prepared by T. Fujita TF Date Prepared 21 January 1986
Participants	**Distribution**
Gerry Tsandoulas, MIT Lincoln Lab Tosh Fujita, JPL	R. Beatty C. Bell H. Bloomfield R. Caputo D. Carlson E. Chow W. Deininger R. Ewell R. Ferber M. Grossman J. Heller L. Isenberg L. Jaffe R. Manvi J. Mondt B. Nesmith T. Newell M. Parker J. Roschke J. Rose J. Spanos J. Stallkamp G. Stapfer J. Stevens J. Stultz V. Truscello S. Voss L. White

BACKGROUND

At the SDAT meeting of 20 January 1986, several ideas were discussed that have ramifications regarding the shape of the antenna. Ideas revolving around the location of the SP-100 in the same plane as the radar antenna were discussed from the perspective of shield weight savings due to the saml1 cross-section of the antenna. The notion of locating the daisy wheel arrangement in the center with the radar forming an outer annulus was mentioned as a possibility. The purpose of the call was to obtain inputs regarding the radar design and performance implications of these arrangements.

ANTENNA SHAPE

Regarding antenna shapes, Dr. Tsandoulas indicated that the annular radar would be unsatisfactory. The radar must be an unbroken surface. The length of the radar normal to the direction of motion is a critical parameter, which is determined from a complex set of trade-offs. The area of the array is then determined. Rectangular shapes are desired and the aspect ratio (length parallel to motion/length perpendicular to the motion) usually ranges from 2:1 to 4:1, where 4:1 is regarded as an upper limit. For very large antennas, aspect ratios of 5:1 may be possible.

CO-PLANAR SP-100 AND RADAR ANTENNA CONCEPTS

Dr. Tsandoulas indicated that a co-planar arrangement was undesirable due to radar scattering effects. He noted that location of the SP-100 behind the plane of the radar was the preferred location.

CONFERENCE REPORT

JET PROPULSION LABORATORY

REPORT NO. TES-354-86-018

SUBJECT Interference Effects on SBRs due to SP-100 Location

PROJECT ______ CONTRACTOR ______ CONTRACT or

ACTION REQUIRED BY ______

TELECON Initiated by T. Fujita CONFERENCE at ______ Date of Occurrence 24 January 1986	Report Prepared by T. Fujita TF Date Prepared 24 January 1986
Participants	**Distribution**
Gerry Tsandoulas, MIT Lincoln Lab Tosh Fujita, JPL	R. Beatty, C. Bell, H. Bloomfield, R. Caputo, D. Carlson, E. Chow, W. Deininger, R. Ewell, R. Ferber, M. Grossman, J. Heller, L. Isenberg, L. Jaffe, R. Manvi, J. Mondt, B. Nesmith, T. Newell, M. Parker, J. Roschke, J. Rose, J. Spanos, J. Stallkamp, G. Stapfer, J. Stevens, J. Stultz, V. Truscello, S. Voss, L. White

BACKGROUND

In an earlier Conference Report (TES-354-86-013 dated 21 January 1986), Dr. Tsandoulas indicated that a co-planar location of the SP-100 with the plane of the phased array was undesirable due to interference effects. Questions were raised at the SDAT meeting of 23 January 1986 regarding the magnitude of the interference and its possible alleviation by shifting the SP-100 so that it would be located in a plane offset by ~1 m from the plane of the phased array. (Rolando Jordan of JPL indicated to Len Jaffe that this offset coupled with shutting off the outer ring of modules would probably be acceptable in terms of reducing interference effects.) The call was placed to Tsandoulas to obtain his views.

INTERFERENCE EFFECTS

Regarding the ~1 m offset suggestion, Dr. Tsandoulas indicated that we should consider this candidate only if it provided significant benefits. The impact on radar performance involves a very complex analysis and is difficult to assess. There are radar signal diffraction effects from the SP-100, which must be handled, and it is desirable from the radar performance point-of-view to locate the SP-100 as far from the phased array plane as possible.

ADDITIONAL INFORMATION

Dynamic Electronic Compensation

This system is envisaged as using small lasers on the outer edge of the phased array to determine flatness characteristics, which will then be used to calibrate the electronic compensation system. The system is not intended to compensate for vibration modes. Instead, the surface flatness will be periodically checked for the purpose of calibrating the electronics. The mass increment is expected to be small.

Mass Differences between the Lincoln Laboratory and Navy NRL Phased Arrays

The Lincoln Laboratory 16 m x 32 m L-Band phased array of area 512 m^2 and mass of 8818 kg has a unit mass of 17.2 kg/m^2. The Navy NRL 15 m x 50 m VHF phased array has an area of 750 m^2, a mass of 8182 kg and a unit mass of 10.9 kg/m^2. Dr. Tsandoulas notes that L-Band and VHF have similar flatness requirements. VHF systems have fewer modules, but these modules are heavier than L-Band modules. Mass differences are probably caused mainly by differences in structural design approaches and requirements for the radar. In order to discern trends with regard to frequency band, comparisons need to be made using a consistent set of design ground rules. Tsandoulas recalls that they did study UHF designs and believes that Darryl Weidler may have some data.

TF:mr

B-2. LOAD REGULATION TIME

J. Stallkamp

May 8, 1986

JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM
#342-86-E-043

TO: Len Jaffe

May 8, 1986

FROM: John Stallkamp John Stallkamp

SUBJECT: SDAT Action Item 36

Load Regulation Time for SBR SRPS
The SRPS load following capability expected fromthe currently proposed implementation can be expected to approach but not formally meet the expressed, idealized, SBR requirement.
Consideration could be given to revising the load following values in the current SRPS specificaion; the proposed implementation can certainly provide a higher response rate.

The present SRPS specification requires a power rate of change of 100 kW per second with a goal of 2 kW per millisecond. Both these rates are slow compared with those expected to be achieved with the proposed implementation using shunt radiators. When asked in early competition both thermoelectric and thermionic contractors indicated that the maximum rates of change would be ultimately limited by the amount of voltage change including overshoot that would be permitted.

A specific proposal by G.E. used parallel switching transistors, RCA type 2N6693, operating in a pulse width modulation mode at 20 Khz with filters to isolate the shunt regulator from the user load. Full load transfer in 10 ms with excellent voltage control can be expected. Faster transfer rates down to a few milliseconds are possible and practical, accompanied with increasing voltage transients. If significantly shorter times are required, the p.w.m. frequency can be increased to a few hundred kilohertz. However different components would be needed and a sensible mass penalty could occur.

It is believed that the rates achievable with the above specific proposal will satisfy many potential users such as electric propulsion. However it does not meet the radar requirement as expressed by MIT.

In a series of phone calls with Lincoln Labs in April 1985, it was stated that the desired time to switch between two pulse repetition intervals (PRI) and re-establish a highly identical repetitive pulse train was 250 microseconds. (Load changes between half and full load are expected, which correspond to changes in PRI over ranges of 2 to 1 with 250 sec the minimum PRI). The above requirement was acknowledged to be the ideal case that would result in no constraints being placed on the operation of the radar. It was also stated that it was not known how much slower a time could be permitted.

In the simplest terms the time to stabilize the power flow at the new rate adds to the time needed to complete a radar search event. The minimum time is the sum of the round trip time to maximum range and the time (PRI X N) where N is the number of reflected pulses from a sequence of very nearly identical transmit pulses that must be received to develop the required statistics. Typical total times range from 0.05 to 0.2 second. Thus a lost time of 5 ms is certainly not desirable but also would not be totally catastrophic. Times of a few milliseconds could be thoroughly acceptable.

Of course, the above simplistic description is far from enough to establish a real requirement. Other people at JPL have been contacted and, as could be expected, the additional information in fact results in more questions.

In conclusion the load following capability of the proposed implementation can be expected to approach but not formally meet the expressed, idealized, SBR requirement. Secondly consideration could be given to revising the load following values in the current SRPS specification; the proposed implementation can certainly provide a higher response rate.

Finally establishing a real SBR requirement may require a face-to-face meeting with appropriate people because many features of the SBR task are classified. This writer has a number of questions to ask and areas that need to be clarified before he would feel he would understand and be able to properly present and properly interpret a requirement statement because of the several trade-offs that are certainly involved.

JS:eh

B-3. SBR SPACECRAFT CONFIGURATIONS AND ANTENNA SUPPORT

J.H. Stevens

February 11, 1986
Revised March and September, 1986

Note: Many of the figures originally included in this document are now in the body of the report, and are not reported here.

JPL

SP-100
SPACE-BASED RADAR (SBR)
CONFIGURATION DESCRIPTION

- ANTENNA ARRAY--32 M x 64 M FLAT
  - 4 QUADRANTS--16 M x 32 M
    16 PANELS/QUADRANT, EACH 2 M x 16 M
  - 64 PANELS TOTAL: 46,800 KG TOTAL

- POWER CONDITIONING EQUIPMENT
  - FROM LINCOLN LABS REPORT
    2 KW UNIT: 6 IN x 8.5 IN x 19.0 IN.
  - SCALING FOR 300 KW AND 64 PANELS
    4.69 KW/PANEL
    ~0.15 M x 0.25 M x 1.0 M PER PANEL (DISTRIBUTED)
  - FROM PWR CONVERSION EFFICIENCY CALCULATIONS BY STALLKAMP
    4.27 KG/KW x 4.69 KW = 20 KG/PANEL; 1,280 KG TOTAL

- SIGNAL PROCESSOR
  - FROM LINCOLN LABS REPORT
    28 MODULES INTEGRATED INTO ONE UNIT
      EACH 7.0 IN X 8.25 IN X 15.25 IN AND 21.4 LB
  - DOES <u>NOT</u> NEED TO BE SCALED UP FOR 32 M x 64 M ARRAY
  - 28 MODULES ARRANGED TO BEST SUIT CONFIGURATION
    ONE PKG AT 0.5 M x 1.0 M x 1.3 M, AND 300 KG

JHS-3
2-11-86

SP-100
SBR MISSION
CONFIGURATION DESIGN
-CONSTRAINTS-

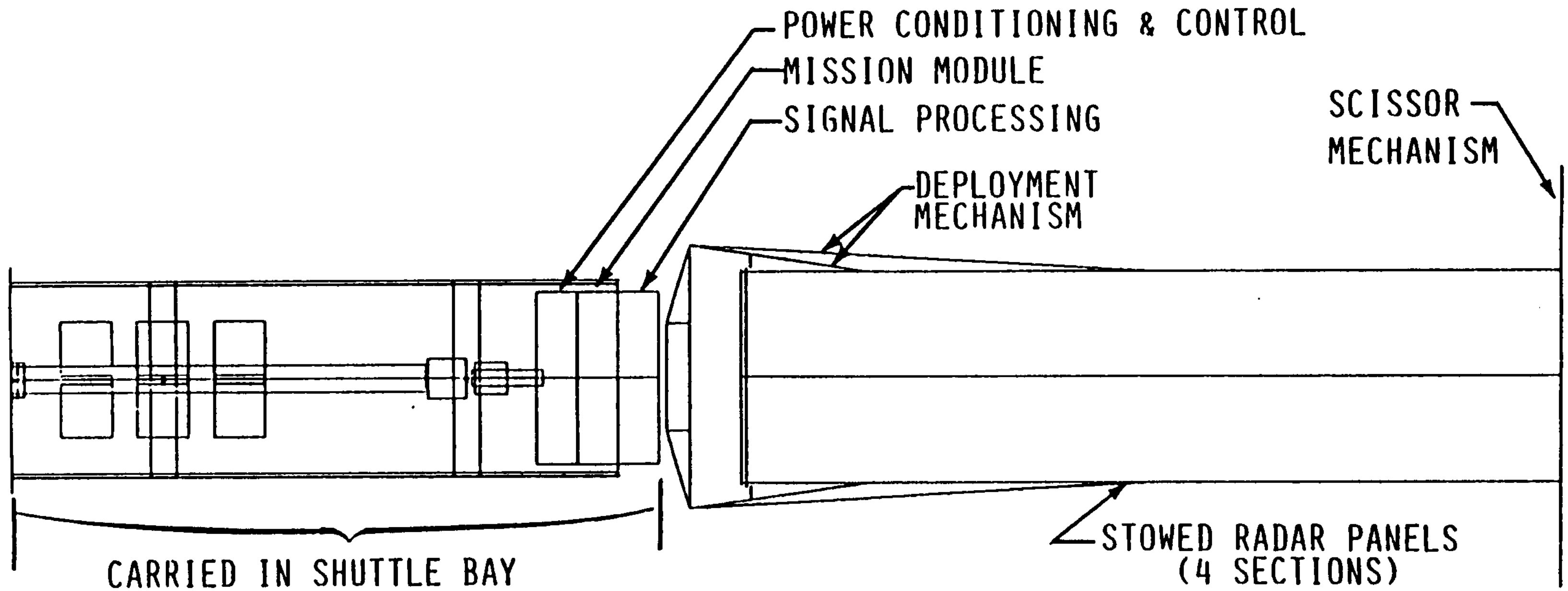
- SBR ARRAY TO BE NORMAL TO NADIR VECTOR
- SBR MAJOR AXIS TO BE ALONG VELOCITY VECTOR
- ARRAY TO BE STABLE AND POINTED TO WITHIN ±0.2 DEGREES
- ARRAY TO BE RIGID AND FLAT TO WITHIN 10 MM (0.4 INCHES)
- STRUCTURE/CONFIGURATION TO PRECLUDE THERMAL DISTORTION
- NO HARDWARE TO BE FORWARD OF GROUND PLANE (OTHER THAN SBR ANTENNAS)
- ARRAY TO BE STOWABLE/DEPLOYABLE
    - STOW COMPACTLY TO MINIMIZE LAUNCHES
    - DEPLOY AUTONOMOUSLY W/O MAN OR OMV ASSIST
    - <u>GOAL:</u> <u>RE</u>STOWABILITY
      TO ALLOW ON-ORBIT SBR CHECK-OUT
- ON-ORBIT ASSEMBLY REQUIRED
    - PRESUMES OMV AND/OR MANNED ASSISTANCE
- ORBIT TRANSFER
    - HIGH THRUST STAGE
      ENTIRE VEHICLE STOWED (SBR AND SP-100)
    - LOW THRUST STAGE
      (COULD BE EXPENDABLE OTV PROP MODULE)
      VEHICLE <u>MAY</u> BE DEPLOYED
        OBVIATES NEED FOR <u>RE</u>STOW
        SIMPLIFIES ON-ORBIT ASSEMBLY
        PROVIDES FOR SIMPLER MORE RELIABLE STRUCTURE/MECHANICS

JPL

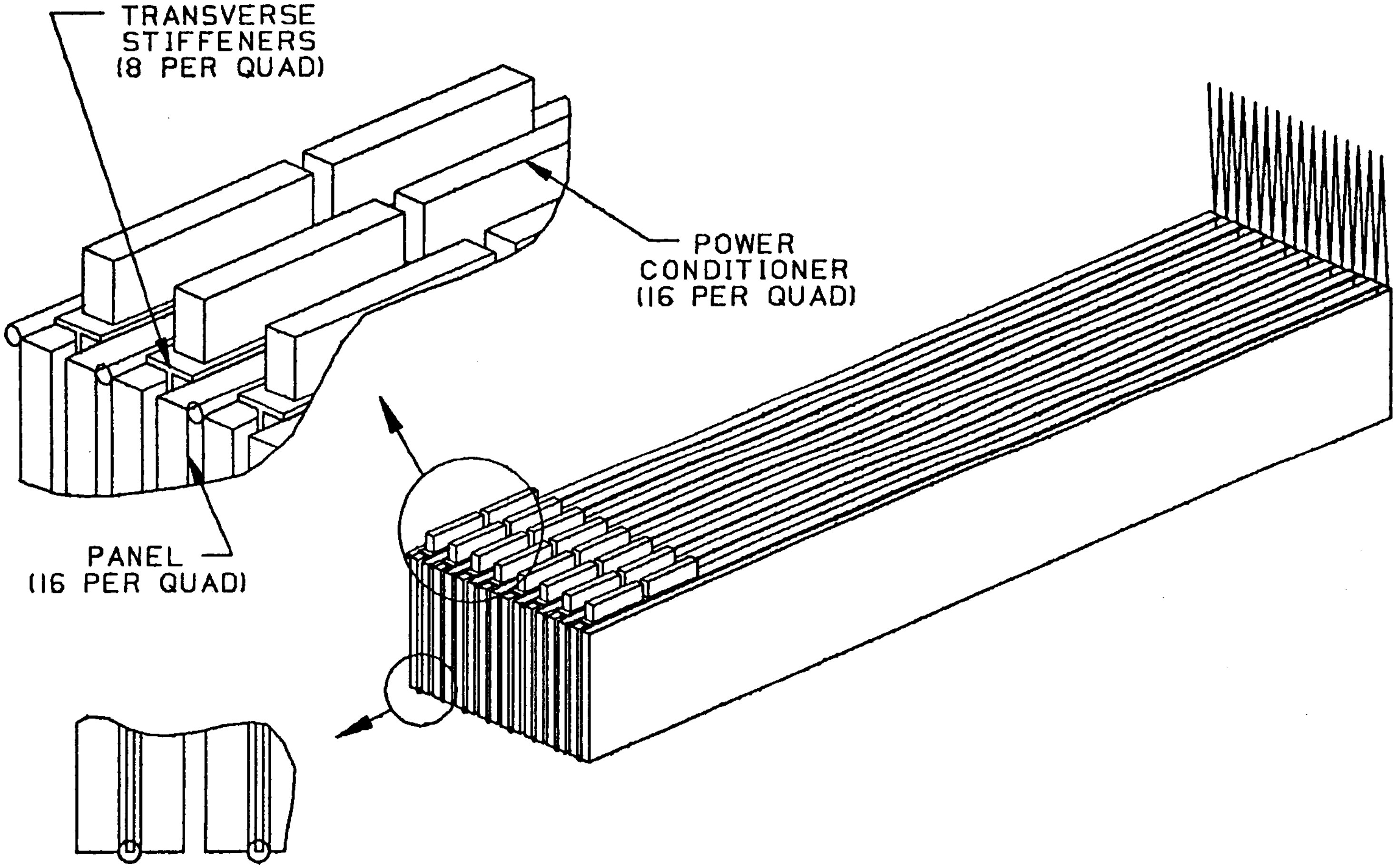
SP-100

SPACE-BASED RADAR MISSION

CONFIGURATION PRIOR TO DEPLOYMENT



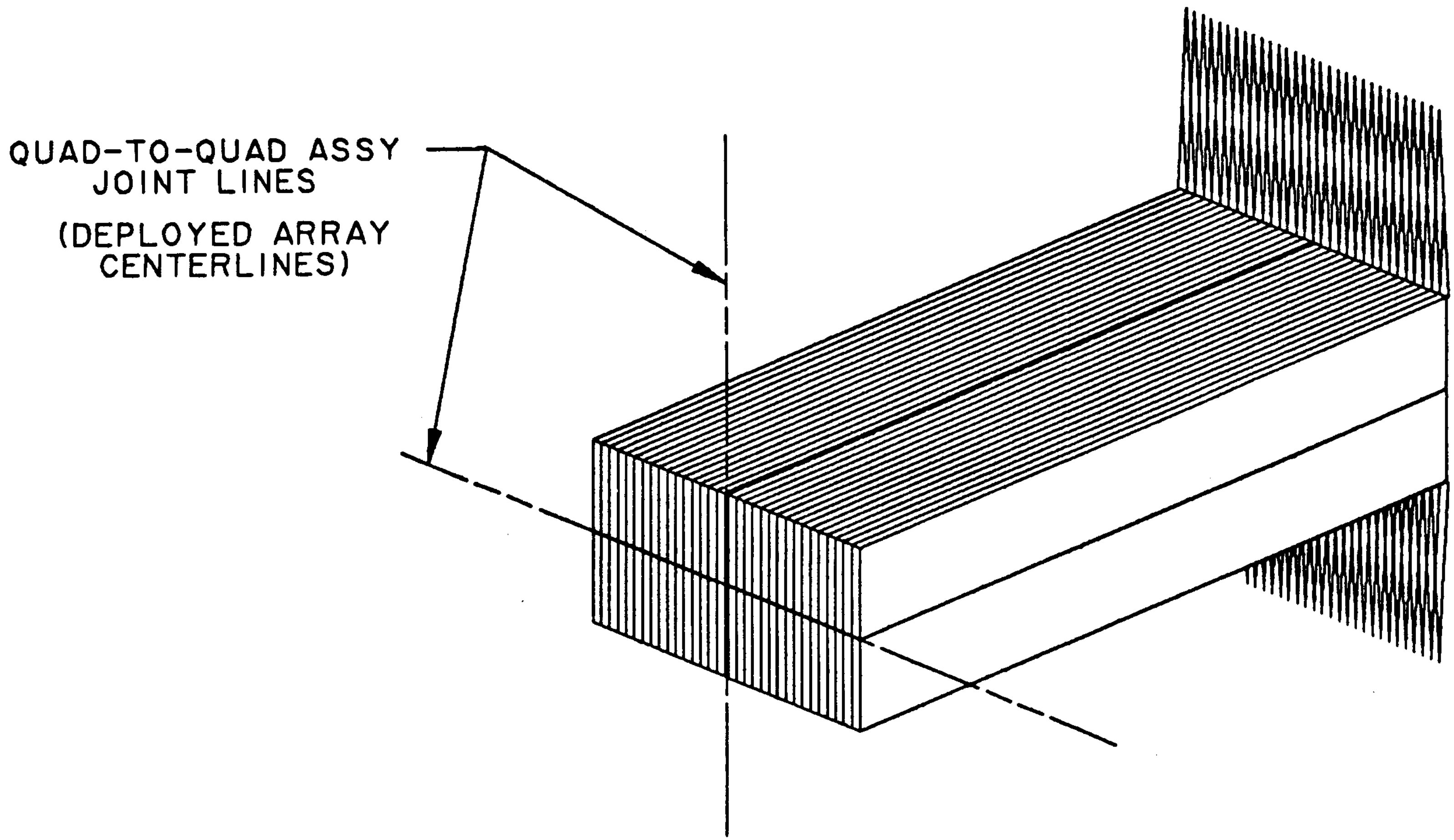
SP-100
SBR MISSION



STOWED ARRAY QUADRANT

JHS - 4g
2-11-86

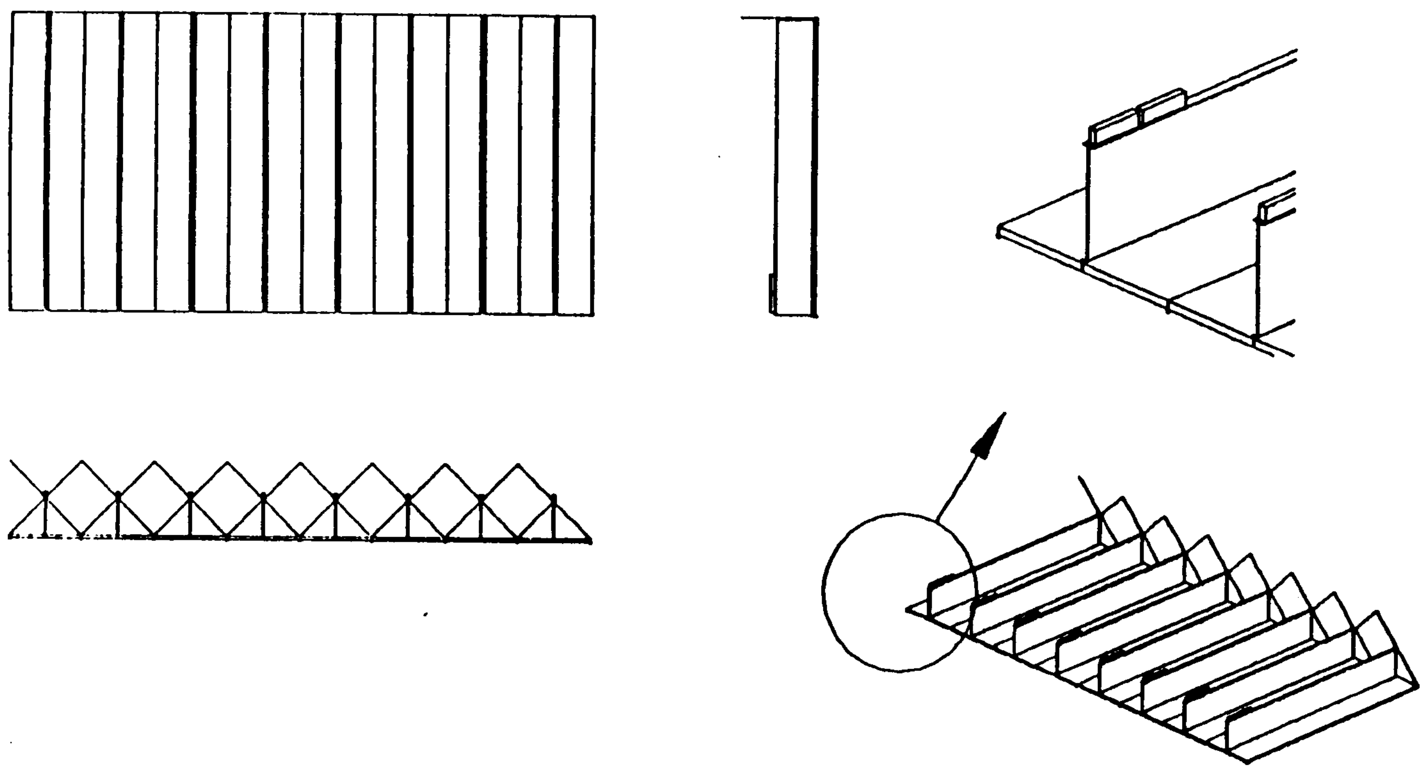
SP-100
SBR MISSION



STOWED ARRAY QUADRANT

JPL

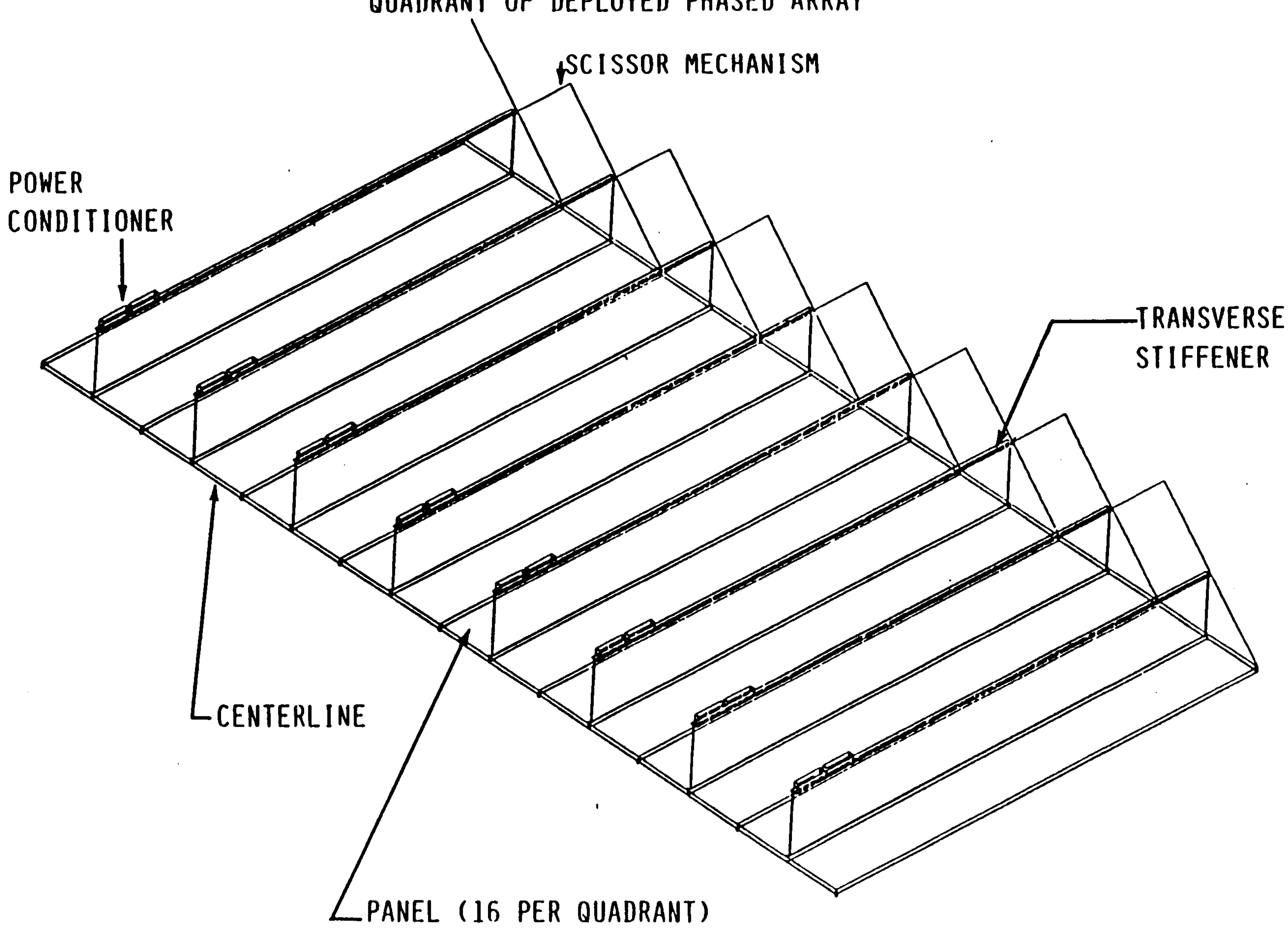
SP-100 SBR MISSION



DEPLOYED ARRAY QUADRANT

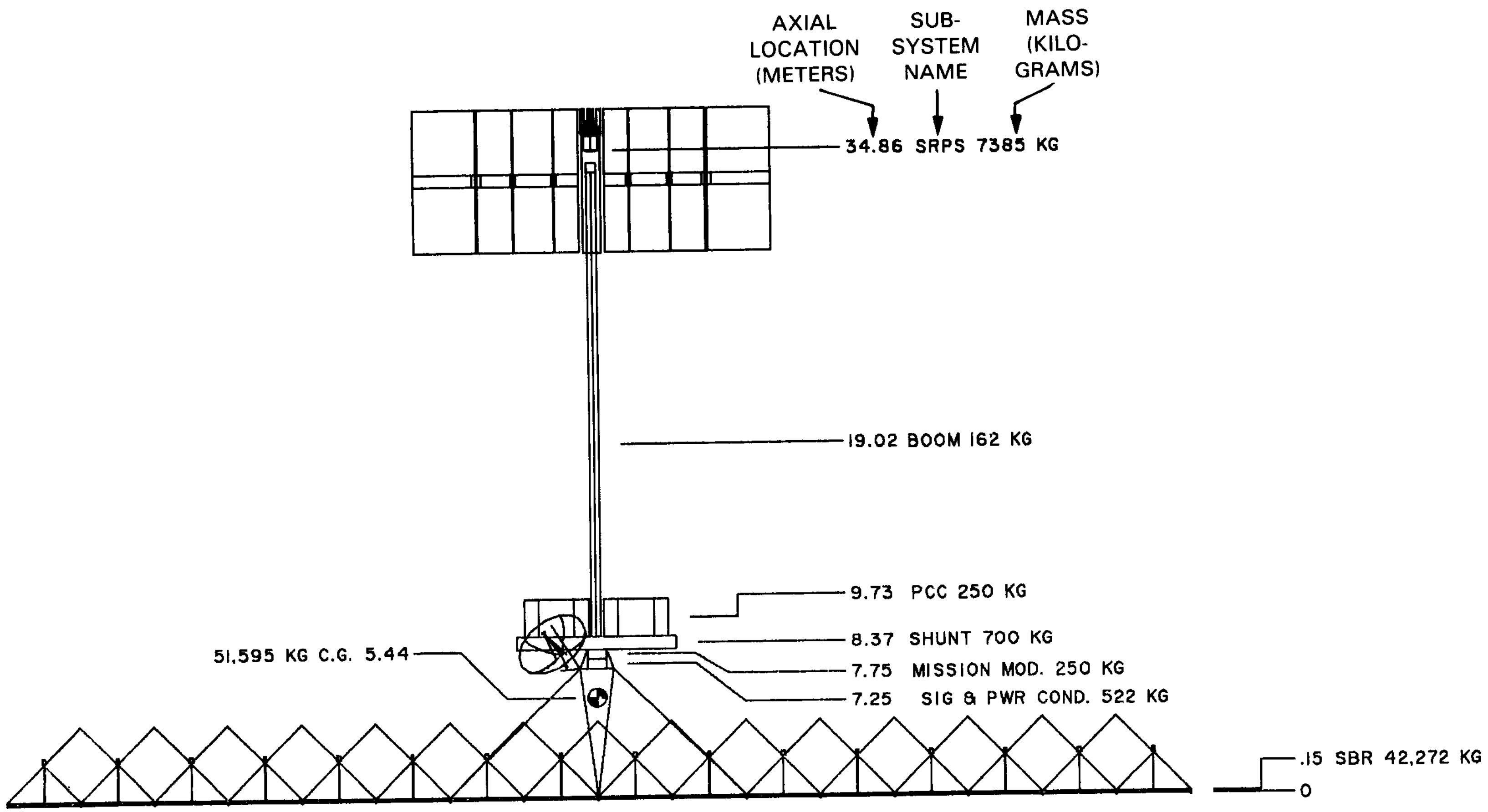
SP-100

SPACE-BASED RADAR MISSION



THE FOLLOWING PAGE SHOWS THE
OTV MASS DISTRIBUTION

OTV MASS DISTRIBUTION



B-4. SHIELD MASS TO REDUCE GAMMA DOSE

L. Jaffe

May 14, 1986

CONFERENCE REPORT

JET PROPULSION LABORATORY

REPORT NO. ____________

Page 1 of 1

SUBJECT Shield mass penalty to reduce gamma dose

PROJECT SP-100 CONTRACTOR ____________ CONTRACT or

ACTION REQUIRED BY ____________

TELECON Initiated by L. Jaffe	Report Prepared by L. Jaffe
CONFERENCE at JPL	Date Prepared May 14, 1986
Date of Occurrence May 14, 1986	
Participants	**Distribution**
Don Carlson - Los Alamos Len Jaffe - JPL	Rose Mondt Truscello Fujita White Stevens Nesmith Carlson Parker Scott

Following up the SDAT recommendation to reduce the gamma fluence delivered to the payload by the SRPS from 5×10^5 rad to 1×10^5 rad, I had asked Don to estimate the additional shielding required.

For a 4.3-m diameter shielded area, using the daisy configuration, he estimated:

At 25-m separation distance: 170 kg
At 50-m separation distance: 130 kg

To shield the 32 x 64 m SBR antenna he estimated that, for a 25-m separation between reactor and antenna, the mass of tungsten would increase 40%. Since the mass calculated for 5×10^5 rad was 1350 kg, of which 66% was tungsten, the mass would be

$$1350 \times 0.66 \times 0.40 - 355 \text{ kg.}$$

(For the same antenna with 50-m separation, the mass of tungsten would increase 56%; the shield for 5×10^5 rad was 61% tungsten.)

Note, however, that our nominal SBR configuration uses 25-m separation between the reactor and <u>user plane</u>, with the antenna 6 m from the user plane. For this configuration, the increased distance provides an additional attenuation of $(31/25)^2 \cong 1.54$. The attenuation to be provided by the shield is then not x 5 but x 5/1.54 = 3.25. By my approximation, the shield mass increase for this configuration is then

$$(3.25/5) * 355 = 230 \text{ kg.}$$

B-5. SHUTTLE CAPABILITIES TO BE ASSUMED

L. Jaffe

May 27, 1986

JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM

27 May 1986

TO: SDAT

FROM: L. Jaffe

SUBJECT: Shuttle Capabilities

SP-100 spacecraft orbital calculations have been difficult to compare because of varying views and varying assumptions as to Shuttle performance.

To resolve these difficulties, T. Fujita, W. Gray, L. White and I have selected a set of assumptions which we judge to entail low to medium risk for our strawman schedules. We have gone over our conclusions with Jack Heller.

The selected capabililities and characteristics are shown on attached sheets. These values should be used for all SP-100 system and mission calculations pertaining to the current strawman missions and to any other missions intended for flight in the 1995-2000 time period.

encls: text
graph

dist: SDAT member & info lists

27 May 1986

SHUTTLE CAPABILITIES TO BE ASSUMED FOR SP-100
(time period 1995-2000)

PERFORMANCE

Mass capability vs altitude is shown on the attached figure for:

Orbital inclinations 28.5 and 57 deg
Main engine thrust levels of 104 and 109%.

These mass capabilities are after deduction so STS manager's and operations reserves. To get the allowable cargo mass, subtract the mass needed for RMS, EVA an ASE from the values shown.

RMS

The mass for 1 RMS is 575 kg. The mass for 2 RMS's is twice this.

ASE

The mass for ASE is to be taken as 6% of the mass of the equipment it supports, except when an ASE design mass is available for the specific equipment to be supported.

EVA

Additional mass required for EVA is to be taken as 100 kg.

OTV MISSION

Main engine thrust is:

109% (if needed) for launch of OTV,
104% for cargo to be carried to GEO.

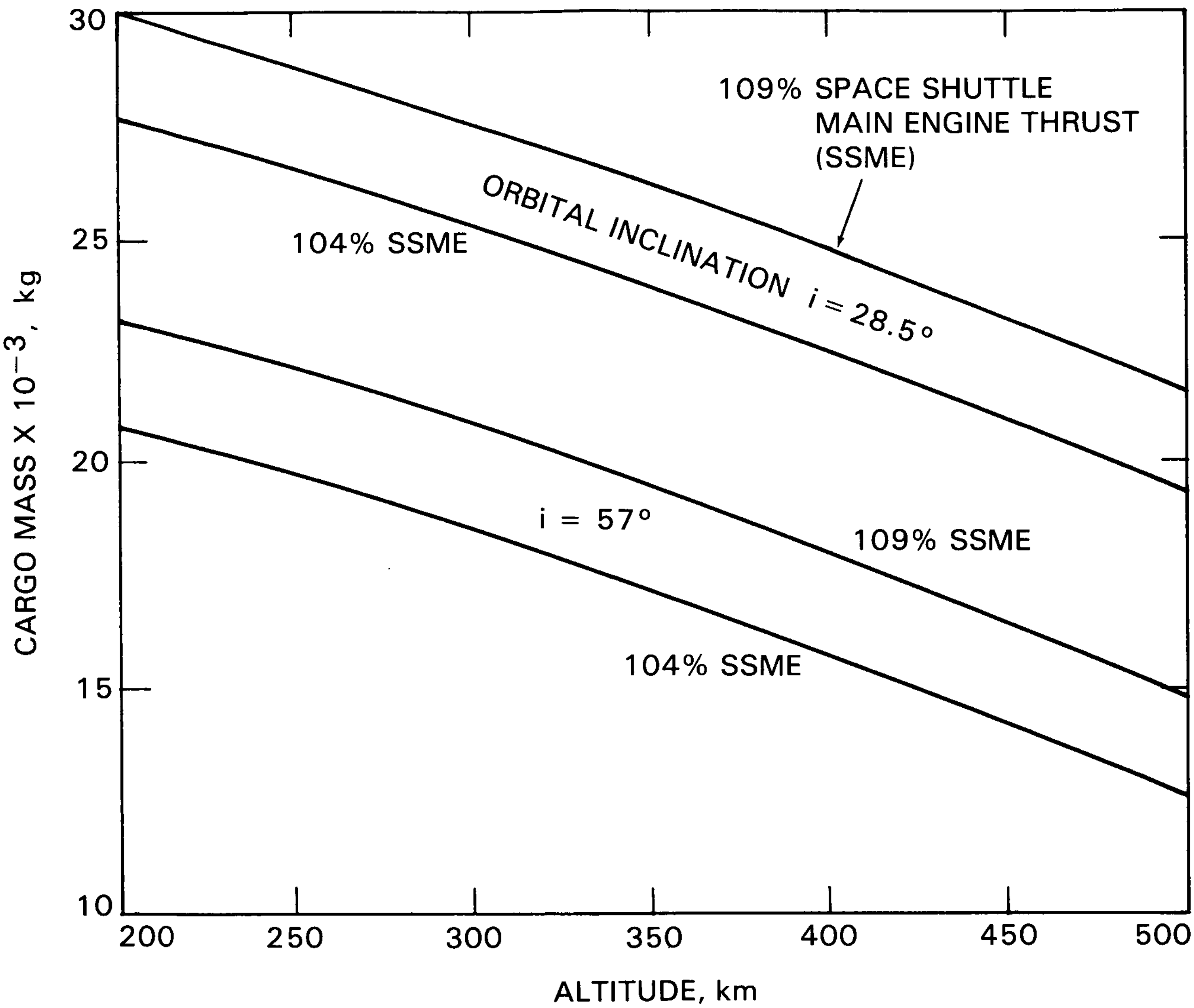
Orbitalassembly will utilize OMV and, if required,EVA. (No RMS.)

SBR MISSION

Main engine thrust is 104%

Orbital assembly will utilize 1 RMS.

Assumed STS Cargo Capability



B-6. PROPULSION PERFORMANCE FOR TRANSFER FROM SHUTTLE ORBIT TO OPERATIONAL ORBIT

T. Fujita
W.B. Gray

February-June 1986

The attached tables were calculated separately from the graphs and were calculated at a later time. Portions not pertinent to the SBR mission are omitted.

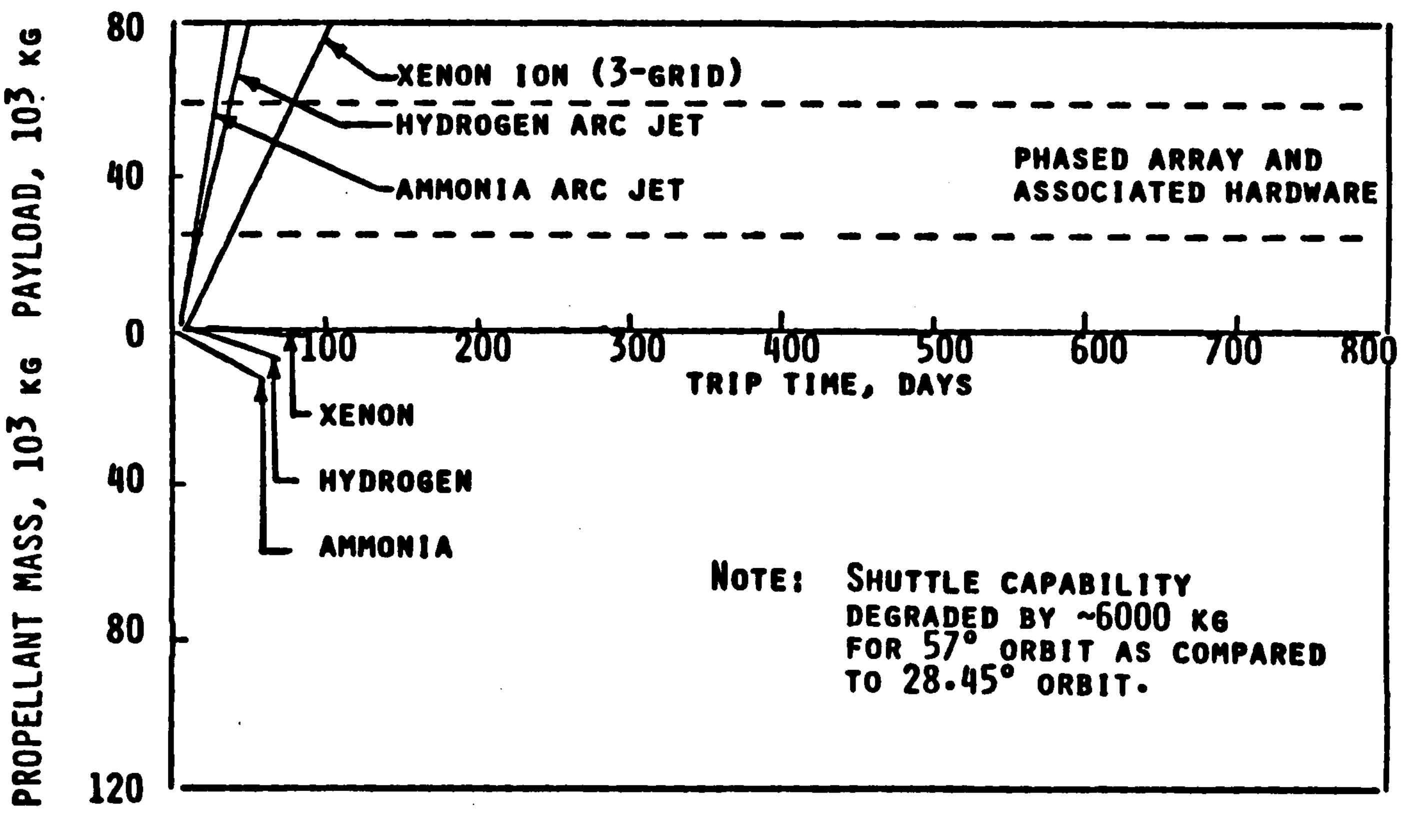
Orbit-raising with on-board electric propulsion

T. Fujita

February 1986

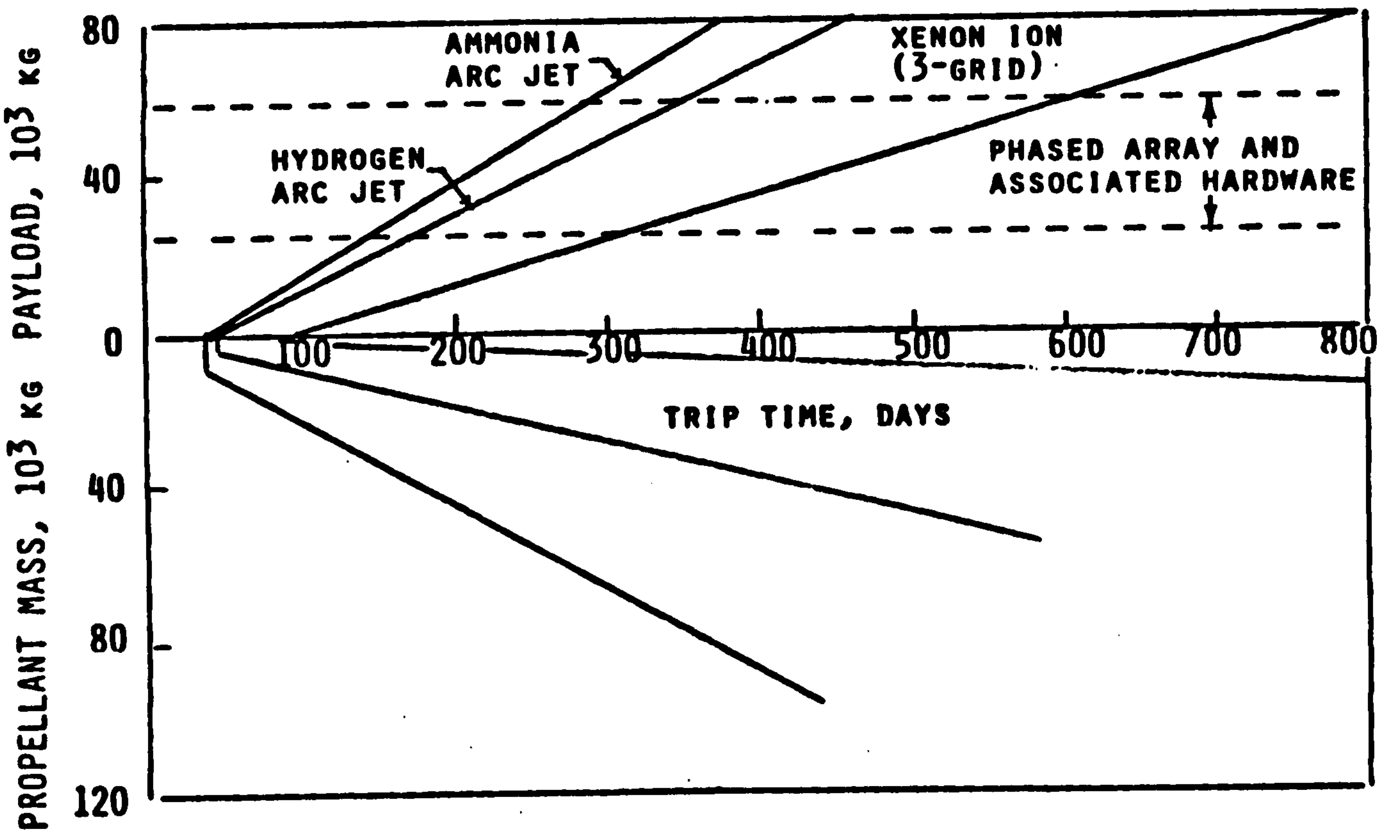
JPL

SPACE-BASED RADAR
ORBIT-RAISING WITH ON-BOARD ELECTRIC PROPULSION
PROJECTED 1993 TECHNOLOGY LEVEL
MISSION: 57°, 278 KM ORBIT TO 61°, 1100 KM ORBIT
POWER LEVEL: 300 KW



JPL

SPACE-BASED RADAR
ORBIT-RAISING WITH ON-BOARD ELECTRIC PROPULSION
PROJECTED 1993 TECHNOLOGY LEVEL
MISSION: 28.45°, 278 KM ORBIT TO 61°, 1100 KM ORBIT
POWER LEVEL: 300 KW



JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM

311.3-1631

25 June 1986

TO: L. D. Jaffe

FROM: W. B. Gray WBG

SUBJECT: SP-100 Performance for SBR and OTV Applications

REFERENCES:

1. Beatty, R. G. G., "OTV and SBR performance with new tankage values," JPL IOM 312/86.5-2422, dated 31 March 1986
2. Gray, W. B., "OTV Performance with Updated Tankage Values," JPL IOM 311.3-1575, dated 17 April 1986
3. Jaffe, L. and Fujita, T., "Electric propulsion characteristics," JPL IOM 313.1.86-15FTHL, dated 20 June 1986
4. Jaffe, L., "Shuttle capabilities," JPL IOM 313.1.86-21FTHL, dated 27 May 1986
5. Gray, W. B., "STS Performance Capability," JPL IOM 311.3-1605, dated 29 May 1986
6. Palaszewski, B., "Hydrogen-, Ammonia- and Xenon-Propellant-Feed Systems," JPL IOM 353PSA-86-098, dated 11 March 1986

Performance parameters for SP-100 powered electric propulsion systems as presented in References 1 and 2 have been updated to reflect the current design options. Propulsion system parameters were taken from Reference 3, STS delivery capability and groundrules from References 4 and 5, and tankage factors from Reference 6.

SBR delivery cases were considered with assembly at a 475 km low earth orbit at either 28.5 or 57 degrees inclination with transfer orbit ignition occuring at 450 km due to orbital decay, and assembly at the Space Station orbit (500 km altitude, 28.5 degree inclination). All of the SBR cases used a nominal 300 kw reactor. Electric propulsion systems consisted of NH_3 arcjets (Isp = 1000 and 1100 seconds) and Xenon ion thrusters (Isp = 2220, 3000, 3684, and 4710 seconds). Cases were generated for all chemical propulsion, all electric propulsion, and a hybrid propulsion system with chemical transfer to an altitude of 925 km and electric propulsion to the operational orbit. Additional cases were run for all electric propulsion from an initial orbit of 420 km altitude and 57 degrees inclination for each thruster system. Ground based propellant tanks were used for all cases, with multiple tanks required due to STS performance limitations. The tanks were sized to provide only the amount of propellant required for the mission. Results of the SBR cases are presented in Attachment 1.

JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM

311.3-1667

4 August 1986

TO: L. D. Jaffe

FROM: W. B. Gray WBG

SUBJECT: Revised SP-100 Performance for SBR

REFERENCE: Gray, W. B., "SP-100 Performance for SBR and OTV Applications," JPL IOM 311.3-1631, dated 25 June 1986

This memo revises the SP-100 performance data for SBR applications as presented in the Reference. The basis for revision is the dry mass of the chemical propulsion system as used for the two-stage and chemical systems. As per our telephone conversation of 4 August, a constant dry mass of 3300 kg was used for the Centaur chemical propulsion system. The attached tables present the revised performance. It should be noted that the all-electric performance is not changed, only the two-stage and chemical performance.

Distribution:

Fujita, T.
Isenberg, L.
O'Toole, R.

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SBR TO ORBIT DELIVERY SCENARIOS
4 August 1986

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START:	LEO/28.5				LEO/57				SPACE STATION			
TRANSFER:	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC
H0:	450	925	450	450	450	925	450	450	500	925	500	500
I0:	28.45	28.45	28.45	28.45	57.00	57.00	57.00	57.00	28.45	28.45	28.45	28.45
R0:	6828	7303	6828	6828	6828	7303	6828	6828	6878	7303	6878	6878
V0:	7644	7392	7644	7644	7644	7392	7644	7644	7616	7392	7616	7616
H1:	925	1100	1100	1100	925	1100	1100	1100	925	1100	1100	1100
I1:	28.45	61.00	61.00	61.00	57.00	61.00	61.00	61.00	28.45	61.00	61.00	61.00
R1:	7303	7478	7478	7478	7303	7478	7478	7478	7303	7478	7478	7478
V1:	7392	7305	7305	7305	7392	7305	7305	7305	7392	7305	7305	7305
ELLIPTICAL TRANSFER ORBIT (COPLANAR):												
DELVOC-E:	127	44	172	172	127	44	172	172	113	44	158	158
VPERIGEE:	7772	7435	7816	7816	7772	7435	7816	7816	7730	7435	7774	7774
VAPOGEE:	7266	7261	7137	7137	7266	7261	7137	7137	7280	7261	7150	7150
DELVE-C1:	125	43	168	168	125	43	168	168	112	43	154	154
TOTHIDELV	253	87	340	340	253	87	340	340	225	87	312	312
LOWTDELV:	253	87	340	340	253	87	340	340	225	87	312	312
ELLIPTICAL TRANSFER ORBIT (NONCOPLANAR):												
DELI1:	.00	.98	2.66	2.66	.00	1.44	1.73	1.73	.00	.98	2.52	2.52
DELI2:	.00	31.57	29.89	29.89	.00	2.56	2.27	2.27	.00	31.57	30.03	30.03
DELVOC-E:	127	134	397	397	127	191	290	290	113	134	373	373
DELVE-C1:	125	3963	3728	3728	125	328	332	332	112	3963	3748	3748
HIGHTDELV	253	4096	4125	4125	253	520	621	621	225	4096	4121	4121
LOWTDELV:	253	6355	6471	6471	253	812	888	888	225	6355	6458	6458
1000s NH3 Arcjets	Note: Ground based cylinderical tanks except LEO-57 ground based spherical											
MSBR:	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000
POWER:	300	300	300	300	300	300	300	300	300	300	300	300
MPOWER:	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000
NETEPPOW:		300		300		300		300		300		300
MEPS:		8881		9042		4018		4053		8445		8584
EPISP:		1000		1000		1000		1000		1000		1000
EPEFFIC:		.43		.43		.43		.43		.43		.43
EPROCK:		.48		.48		.08		.09		.48		.48
MCHEMDRY:	3300		3300		3300		3300		3300		3300	
CHEMISP:	444		444		444		444		444		444	
CHEMROCK:	.06		.61		.06		.13		.05		.61	
EXNDNT:		63881		64042		59018		59053		63445		63584
EPPROP:		77762		80349		5235		5768		73630		75811
EPTANK:		21548		22057		1709		1847		17424		17879
MEXNEO:		163210		166448		65963		66669		154499		157274
FLOW:		.0026858		.0026858		.0026858		.0026858		.0026858		.0026858
EPTIME:		335.19		346.25		22.56		24.86		317.30		326.70
MEP:		108210		111448		10963		11669		99499		102274
EXLDNT:	166510		58300		69263		58300		157799		58300	
CHEMPROP:	9939		91912		4134		8933		8355		91756	
MCHEM:	13239		95212		7434		12233		11655		95056	
M0:	176449		150212	166448	73397		67233	66669	166155		150056	157274
MPROPT:	121449		95212	111448	18397		12233	11669	111155		95056	102274
STS DELY:	18941		18941	18941	12549		12549	12549	18236		18236	18236
PROP STS:	6.41		5.03	5.88	1.47		.97	.93	6.10		5.21	5.61

START:	LEO/28.5				LEO/57				SPACE STATION			
TRANSFER:	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC
H0:	450	925	450	450	450	925	450	450	500	925	500	500
I0:	28.45	28.45	28.45	28.45	57.00	57.00	57.00	57.00	28.45	28.45	28.45	28.45
R0:	6828	7303	6828	6828	6828	7303	6828	6828	6878	7303	6878	6878
V0:	7644	7392	7644	7644	7644	7392	7644	7644	7616	7392	7616	7616
H1:	925	1100	1100	1100	925	1100	1100	1100	925	1100	1100	1100
I1:	28.45	61.00	61.00	61.00	57.00	61.00	61.00	61.00	28.45	61.00	61.00	61.00
R1:	7303	7478	7478	7478	7303	7478	7478	7478	7303	7478	7478	7478
V1:	7392	7305	7305	7305	7392	7305	7305	7305	7392	7305	7305	7305
ELLIPTICAL TRANSFER ORBIT (COPLANAR):												
DELVOC-E:	127	44	172	172	127	44	172	172	113	44	158	158
VPERIGEE:	7772	7435	7816	7816	7772	7435	7816	7816	7730	7435	7774	7774
VAPOGEE:	7266	7261	7137	7137	7266	7261	7137	7137	7280	7261	7150	7150
DELVE-C1:	125	43	168	168	125	43	168	168	112	43	154	154
TOTHIDELV	253	87	340	340	253	87	340	340	225	87	312	312
LOWTDELV:	253	87	340	340	253	87	340	340	225	87	312	312
ELLIPTICAL TRANSFER ORBIT (NONCOPLANAR):												
DELI1:	.00	.98	2.66	2.66	.00	1.44	1.73	1.73	.00	.98	2.52	2.52
DELI2:	.00	31.57	29.89	29.89	.00	2.56	2.27	2.27	.00	31.57	30.03	30.03
DELVOC-E:	127	134	397	397	127	191	290	290	113	134	373	373
DELVE-C1:	125	3963	3728	3728	125	328	332	332	112	3963	3748	3748
HIGHTDELV	253	4096	4125	4125	253	520	621	621	225	4096	4121	4121
LOWTDELV:	253	6355	6471	6471	253	812	888	888	225	6355	6458	6458

1100s NH3 Arcjets   Note: Ground based cylinderical tanks except LEO-57 ground based spherical

MSBR:	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000
POWER:	300	300	300	300	300	300	300	300	300	300	300	300
MPOWER:	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000
NETEPPOW:		300		300		300		300		300		300
MEPS:		8014		8141		3984		4016		7710		7821
EPISP:		1100		1100		1100		1100		1100		1100
EPEFFIC:		.43		.43		.43		.43		.43		.43
EPROCK:		.44		.45		.07		.08		.44		.45
MCHEMDRY:	3300		3300		3300		3300		3300		3300	
CHEMISP:	444		444		444		444		444		444	
CHEMROCK:	.06		.61		.06		.13		.05		.61	
EXNDNT:		63014		63141		58984		59016		62710		62821
EPPROP:		64594		66901		4728		5206		62194		63920
EPTANK:		17972		18369		1577		1702		14906		15270
MEXNSD:		145879		148411		65289		65924		139810		142011
FLOW:		.0022197		.0022197		.0022197		.0022197		.0022197		.0022197
EPTIME:		338.38		348.84		24.65		27.15		324.30		333.30
MEP:		90879		93411		10289		10924		84810		87011
EXLDNT:	149179		58300		68589		58300		143110		58300	
CHEMPROP:	8905		91912		4094		8933		7578		91756	
MCHEM:	12205		95212		7394		12233		10878		95056	
M0:	158084		150212	148411	72684		67233	65924	150687		150056	142011
MPROPT:	103084		95212	93411	17684		12233	10924	95687		95056	87011
STS DELY:	18941		18941	18941	12549		12549	12549	18236		18236	18236
PROP STS:	5.44		5.03	4.93	1.41		.97	.87	5.25		5.21	4.77

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SBR TO ORBIT DELIVERY [illegible]108
4 August 1986

ORIGINAL PAGE IS OF POOR QUALITY

START:	LEO/28.5				LEO/57				SPACE STATION			
TRANSFER:	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC
H0:	450	925	450	450	450	925	450	450	500	925	500	500
I0:	28.45	28.45	28.45	28.45	57.00	57.00	57.00	57.00	28.45	28.45	28.45	28.45
R0:	6828	7303	6828	6828	6828	7303	6828	6828	6878	7303	6878	6878
V0:	7644	7392	7644	7644	7644	7392	7644	7644	7616	7392	7616	7616
H1:	925	1100	1100	1100	925	1100	1100	1100	925	1100	1100	1100
I1:	28.45	61.00	61.00	61.00	57.00	61.00	61.00	61.00	28.45	61.00	61.00	61.00
R1:	7303	7478	7478	7478	7303	7478	7478	7478	7303	7478	7478	7478
V1:	7392	7305	7305	7305	7392	7305	7305	7305	7392	7305	7305	7305
ELLIPTICAL TRANSFER ORBIT (COPLANAR):												
DELVOC-E:	127	44	172	172	127	44	172	172	113	44	158	158
VPERIGEE:	7772	7435	7816	7816	7772	7435	7816	7816	7730	7435	7774	7774
VAPOGEE:	7266	7261	7137	7137	7266	7261	7137	7137	7280	7261	7150	7150
DELVE-C1:	125	43	168	168	125	43	168	168	112	43	154	154
TOTHIDELV	253	87	340	340	253	87	340	340	225	87	312	312
LOWTDELV:	253	87	340	340	253	87	340	340	225	87	312	312
ELLIPTICAL TRANSFER ORBIT (NONCOPLANAR):												
DELI1:	.00	.98	2.66	2.66	.00	1.44	1.73	1.73	.00	.98	2.52	2.52
DELI2:	.00	31.57	29.89	29.89	.00	2.56	2.27	2.27	.00	31.57	30.03	30.03
DELVOC-E:	127	134	397	397	127	191	290	290	113	134	373	373
DELVE-C1:	125	3963	3728	3728	125	328	332	332	112	3963	3748	3748
HIGHTDELV	253	4096	4125	4125	253	520	621	621	225	4096	4121	4121
LOWTDELV:	253	6355	6471	6471	253	812	888	888	225	6355	6458	6458
2220s Xe Ion thrusters Note: Ground based tanks												
MSBR:	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000
POWER:	300	300	300	300	300	300	300	300	300	300	300	300
MPOWER:	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000
NETEPPOW:		299		299		299		299		299		299
MEPS:		5934		5961		4853		4866		5934		5958
EPISP:		2220		2220		2220		2220		2220		2220
EPEFFIC:		.57		.57		.57		.57		.57		.57
EPROCK:		.25		.26		.04		.04		.25		.26
MCHEMDRY:	3300		3300		3300		3300		3300		3300	
CHEMISP:	444		444		444		444		444		444	
CHEMROCK:	.06		.61		.06		.13		.05		.61	
EXNDNT:		60934		60961		59853		59866		60934		60958
EPPROP:		21246		21719		2281		2502		21246		21665
EPTANK:		1823		1860		250		269		1823		1856
MEXNSD:		84006		84540		62385		62636		84006		84479
FLOW:		.0007200		.0007200		.0007200		.0007200		.0007200		.0007200
EPTIME:		341.57		349.14		36.67		40.22		341.57		348.28
MEP:		29006		29540		7385		7636		29006		29479
EXLDNT:	87306		58300		65685		58300		87306		58300	
CHEMPROP:	5211		91912		3921		8933		4623		91756	
MCHEM:	8511		95212		7221		12233		7923		95056	
M0:	92517		150212	84540	69606		67233	62636	91928		150056	84479
MPROPT:	37517		95212	29540	14606		12233	7636	36928		95056	29479
STS DELY:	18941		18941	18941	12549		12549	12549	18236		18236	18236
PROP STS:	1.98		5.03	1.56	1.16		.97	.61	2.03		5.21	1.62

ORIGINAL PAGE IS
OF POOR QUALITY

START:	LEO/28.5				LEO/57				SPACE STATION			
TRANSFER:	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC
H0:	450	925	450	450	450	925	450	450	500	925	500	500
I0:	28.45	28.45	28.45	28.45	57.00	57.00	57.00	57.00	28.45	28.45	28.45	28.45
R0:	6828	7303	6828	6828	6828	7303	6828	6828	6878	7303	6878	6878
V0:	7644	7392	7644	7644	7644	7392	7644	7644	7616	7392	7616	7616
H1:	925	1100	1100	1100	925	1100	1100	1100	925	1100	1100	1100
I1:	28.45	61.00	61.00	61.00	57.00	61.00	61.00	61.00	28.45	61.00	61.00	61.00
R1:	7303	7478	7478	7478	7303	7478	7478	7478	7303	7478	7478	7478
V1:	7392	7305	7305	7305	7392	7305	7305	7305	7392	7305	7305	7305
ELLIPTICAL TRANSFER ORBIT (COPLANAR):												
DELVOC-E:	127	44	172	172	127	44	172	172	113	44	158	158
VPERIGEE:	7772	7435	7816	7816	7772	7435	7816	7816	7730	7435	7774	7774
VAPOGEE:	7266	7261	7137	7137	7266	7261	7137	7137	7280	7261	7150	7150
DELVE-C1:	125	43	168	168	125	43	168	168	112	43	154	154
TOTHIDELV	253	87	340	340	253	87	340	340	225	87	312	312
LOWTDELV:	253	87	340	340	253	87	340	340	225	87	312	312
ELLIPTICAL TRANSFER ORBIT (NONCOPLANAR):												
DELI1:	.00	.98	2.66	2.66	.00	1.44	1.73	1.73	.00	.98	2.52	2.52
DELI2:	.00	31.57	29.89	29.89	.00	2.56	2.27	2.27	.00	31.57	30.03	30.03
DELVOC-E:	127	134	397	397	127	191	290	290	113	134	373	373
DELVE-C1:	125	3963	3728	3728	125	328	332	332	112	3963	3748	3748
HIGHTDELV	253	4096	4125	4125	253	520	621	621	225	4096	4121	4121
LOWTDELV:	253	6355	6471	6471	253	812	388	888	225	6355	6458	6458

3000s Xe Ion thrusters Note: Ground based tanks

MSBR:	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000
POWER:	300	300	300	300	300	300	300	300	300	300	300	300
MPOWER:	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000
NETEPPOW:		285		285		285		285		285		285
MEPS:		5149		5167		4403		4412		5149		5165
EPISP:		3000		3000		3000		3000		3000		3000
EPEFFIC:		.60		.60		.60		.60		.60		.60
EPROCK:		.19		.20		.03		.03		.19		.20
MCHEMDRY:	3300		3300		3300		3300		3300		3300	
CHEMISP:	444		444		444		444		444		444	
CHEMROCK:	.06		.61		.06		.13		.05		.61	
EXNDNT:		60149		60167		59403		59412		60149		60165
EPPROP:		14789		15101		1666		1826		14789		15065
EPTANK:		1259		1284		199		212		1259		1281
MEXNSD:		76198		76552		61267		61450		76198		76512
FLOW:		.0003956		.0003956		.0003956		.0003956		.0003956		.0003956
EPTIME:		432.72		441.83		48.74		53.42		432.72		440.79
MEP:		21198		21552		6267		6450		21198		21512
EXLDNT:	79498		58300		64567		58300		79498		58300	
CHEMPROP:	4745		91912		3854		8933		4209		91756	
MCHEM:	8045		95212		7154		12233		7509		95056	
M0:	84243		150212	76552	68422		67233	61450	83708		150056	76512
MPROPT:	29243		95212	21552	13422		12233	6450	28708		95056	21512
STS DELY:	18941		18941	18941	12549		12549	12549	18236		18236	18236
PROP STS:	1.54		5.03	1.14	1.07		.97	.51	1.57		5.21	1.18

SBR TO ORBIT DELIVERY SCENARIOS
4 August 1986

ORIGINAL PAGE IS
OF POOR QUALITY

START:	LEO/28.5				LEO/57				SPACE STATION			
TRANSFER:	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC
H0:	450	925	450	450	450	925	450	450	500	925	500	500
I0:	28.45	28.45	28.45	28.45	57.00	57.00	57.00	57.00	28.45	28.45	28.45	28.45
R0:	6828	7303	6828	6828	6828	7303	6828	6828	6878	7303	6878	6878
V0:	7644	7392	7644	7644	7644	7392	7644	7644	7616	7392	7616	7616
H1:	925	1100	1100	1100	925	1100	1100	1100	925	1100	1100	1100
I1:	28.45	61.00	61.00	61.00	57.00	61.00	61.00	61.00	28.45	61.00	61.00	61.00
R1:	7303	7478	7478	7478	7303	7478	7478	7478	7303	7478	7478	7478
V1:	7392	7305	7305	7305	7392	7305	7305	7305	7392	7305	7305	7305
ELLIPTICAL TRANSFER ORBIT (COPLANAR):												
DELVOC-E:	127	44	172	172	127	44	172	172	113	44	158	158
VPERIGEE:	7772	7435	7816	7816	7772	7435	7816	7816	7730	7435	7774	7774
VAPOGEE:	7266	7261	7137	7137	7266	7261	7137	7137	7280	7261	7150	7150
DELVE-C1:	125	43	168	168	125	43	168	168	112	43	154	154
TOTHIDELV	253	87	340	340	253	87	340	340	225	87	312	312
LOWTDELV:	253	87	340	340	253	87	340	340	225	87	312	312
ELLIPTICAL TRANSFER ORBIT (NONCOPLANAR):												
DELI1:	.00	.98	2.66	2.66	.00	1.44	1.73	1.73	.00	.98	2.52	2.52
DELI2:	.00	31.57	29.89	29.89	.00	2.56	2.27	2.27	.00	31.57	30.03	30.03
DELVOC-E:	127	134	397	397	127	191	290	290	113	134	373	373
DELVE-C1:	125	3963	3728	3728	125	328	332	332	112	3963	3748	3748
HIGHTDELV	253	4096	4125	4125	253	520	621	621	225	4096	4121	4121
LOWTDELV:	253	6355	6471	6471	253	812	888	888	225	6355	6458	6458

3684s Ion thrusters Note: Ground based tanks

MSBR:	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000
POWER:	300	300	300	300	300	300	300	300	300	300	300	300
MPOWER:	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000
NETEPPOW:		290		290		290		290		290		290
MEPS:		4753		4766		4165		4173		4753		4765
EPISP:		3684		3684		3684		3684		3684		3684
EPEFFIC:		.69		.69		.69		.69		.69		.69
EPROCK:		.16		.16		.02		.02		.16		.16
MCHEMDRY:	3300		3300		3300		3300		3300		3300	
CHEMISP:	444		444		444		444		444		444	
CHEMROCK:	.06		.61		.06		.13		.05		.61	
EXNDNT:		59753		59766		59165		59173		59753		59765
EPPROP:		11673		11912		1347		1476		11673		11885
EPTANK:		1011		1030		172		183		1011		1028
MEXNED:		72437		72709		60684		60831		72437		72678
FLOW:		.0003070		.0003070		.0003070		.0003070		.0003070		.0003070
EPTIME:		440.13		449.15		50.79		55.65		440.13		446.13
MEP:		17437		17709		5684		5831		17437		17678
EXLDNT:	75737		58300		63984		58300		75737		58300	
CHEMPROP:	4521		91912		3819		8933		4010		91756	
MCHEM:	7821		95212		7119		12233		7310		95056	
M0:	80257		150212	72709	67803		67233	60831	79747		150056	72678
MPROPT:	25257		95212	17709	12803		12233	5831	24747		95056	17678
STS DELY:	18941		18941	18941	12549		12549	12549	18236		18236	18236
PROP STS:	1.33		5.03	.93	1.02		.97	.46	1.36		5.21	.97

SBR TO ORBIT DELIVERY SCENARIOS
4 August 1986

ORIGINAL PAGE IS OF POOR QUALITY

START:	LEO/28.5				LEO/57				SPACE STATION			
TRANSFER:	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC	TWO-STAGE		CHEM	ELEC
H0:	450	925	450	450	450	925	450	450	500	925	500	500
I0:	28.45	28.45	28.45	28.45	57.00	57.00	57.00	57.00	28.45	28.45	28.45	28.45
R0:	6828	7303	6828	6825	6829	7303	6828	6828	6578	7303	6878	6878
V0:	7644	7392	7644	7644	7644	7392	7644	7644	7616	7392	7616	7616
H1:	925	1100	1100	1100	925	1100	1100	1100	925	1100	1100	1100
I1:	28.45	61.00	61.00	61.00	57.00	61.00	61.00	61.00	28.45	61.00	61.00	61.00
R1:	7303	7478	7478	7478	7303	7478	7478	7478	7303	7478	7478	7478
V1:	7392	7305	7305	7305	7392	7305	7305	7305	7392	7305	7305	7305
ELLIPTICAL TRANSFER ORBIT (COPLANAR):												
DELVOC-E:	127	44	172	172	127	44	172	172	113	44	158	158
VPERIGEE:	7772	7435	7816	7816	7772	7435	7816	7816	7730	7435	7774	7774
VAPOGEE:	7266	7261	7137	7137	7266	7261	7137	7137	7280	7261	7150	7150
DELVE-C1:	125	43	168	168	125	43	168	168	112	43	154	154
TOTHIDELV	253	87	340	340	253	87	340	340	225	87	312	312
LOWTDELV:	253	87	340	340	253	87	340	340	225	87	312	312
ELLIPTICAL TRANSFER ORBIT (NONCOPLANAR):												
DELI1:	.00	.98	2.66	2.66	.00	1.44	1.73	1.73	.00	.98	2.52	2.52
DELI2:	.00	31.57	29.89	29.89	.00	2.36	2.27	2.27	.00	31.57	30.03	30.03
DELVOC-E:	127	134	397	397	127	191	290	290	113	134	373	373
DELVE-C1:	125	3963	3728	3728	125	328	332	332	112	3963	3748	3748
HIGHTDELV	253	4096	4125	4125	253	520	621	621	225	4096	4121	4121
LOWTDELV:	253	6355	6471	6471	253	812	888	888	225	6355	6458	6458
4710s Xe Ion thrusters Note: Ground based tanks												
MSBR:	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000	50000
POWER:	300	300	300	300	300	300	300	300	300	300	300	300
MPOWER:	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000	5000
NETEFPOW:		270		270		270		270		270		270
MEPS:		4369		4379		3924		3930		4369		4378
EPISP:		4710		4710		4710		4710		4710		4710
EPEFFIC:		.73		.73		.73		.73		.73		.73
EPROCK:		.13		.13		.02		.02		.13		.13
MCHEMDRY:	3300		3300		3300		3300		3300		3300	
CHEMISP:	444		444		444		444		444		444	
CHEMROCK:	.06		.61		.06		.13		.05		.61	
EXNDNT:		59369		59379		58924		58930		59369		59378
EPPROP:		8863		9040		1046		1146		8863		9020
EPTANK:		786		800		146		155		786		798
MEXNSO:		69018		69219		60116		60231		69018		69196
FLOW:		.0001850		.0001850		.0001850		.0001850		.0001850		.0001850
EPTIME:		554.55		565.63		65.46		71.71		554.55		564.37
MEP:		14018		14219		5116		5231		14018		14196
EXLDNT:	72318		58300		63416		58300		72318		58300	
CHEMPROP:	4317		91912		3785		8933		3829		91756	
MCHEM:	7617		95212		7085		12233		7129		95056	
M0:	76634		150212	69219	67202		67233	60231	76147		150056	69196
MPROPT:	21634		95212	14219	12202		12233	5231	21147		95056	14196
STS DELY:	18941		18941	18941	12549		12549	12549	18236		18236	18236
PROP STS:	1.14		5.03	.75	.97		.97	.42	1.16		5.21	.78

B-7. MINIMUM ALTITUDE FOR SPACECRAFT ASSEMBLY. ORBITAL DECAY, AND ORBITAL LIFETIME

L. Jaffe

June-September 1986

The attached memos deal with orbital decay and orbital lifetime of three different configurations of the spacecraft:

1) Individual elements of the spacecraft, parked in orbit prior to assembly.

2) Completed spacecraft, assembled, but not deployed.

3) Complete spacecraft, fully deployed (during electric propulsion).

In reading these memos, it is important to note which of these configurations is considered in each memo.

In the September 16 memo, Figure 1 has been removed; as it is included in the body of this report as Figure 13-1.

JET PROPULSION LABORATORY INTEROFFICE MEMORANDUM

September 16, 1986

SUBJECT: Minimal altitude for SBR assembly

FROM: L. Jaffe

TO: SDAT and Information List

Ref. a: IOM 313.1-86-16, this subject, Jaffe to SDAT

b: IOM 313.1-86-030FTHL, "Orbital Decay Rate during SBR Assembly", Jaffe to SDAT, 7/16/86

c: IOM 313.1-86-036, "Minimal Altitude and Orbital Decay Rate during SBR Assembly", Jaffe to SDAT, 8/4/86

This is an update of ref. a, giving the minimum altitude for SBR orbital assembly. The primary change is in expanding the information on orbital lifetime vs. altitude for 2 antenna quadrants, Table 2.

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SUMMARY

The minimum altitude is set by the requirement that elements of the SBR spacecraft, left in orbit by the multiple Shuttle sorties needed, must not decay into the atmosphere before assembly is completed. The recommended orbital lifetime is 1 year.

The altitude to insure 1 year orbital lifetime for various possible SBR orbiting elements is shown in Table 1 for various cases. A recommended minimum altitude is included for each object and is based on the discussion of attitude, below. For the scenarios investigated so far, the assembly altitude required is about 400 km.

J. Heller and H. Bloomfield have proposed scenarios using assembly at 200-278 km. Table 2 gives the orbital lifetime over a range of altitudes for a pair of SBR antenna quadrants, with and without their supporting structure. At 200 km, the orbital lifetime is only about 5 days; at 278 km, it is about 30 days.

ASSUMPTIONS AND DATA SOURCES

Attitude

The attitude the SBR elements will assume without active attitude control was estimated. Conversations with P. Jaffe and T. Kia of JPL indicated that at the altitudes of interest the major external disturbing torque is due to aerodynamic drag. For the objects considered, the long axis will tend to orient along to the velocity vector but will oscillate and may tumble. The objects will also tend to spin about their long axis.

The tables gives orbital decay results for 6 and 90 degree angles of attack (angles between the velocity vector and the long axis of the element). I suggest use of a weighted average (weighted 2:1) as a conservative estimate to cover the uncertainty in projected area arising from the oscillations. These weighted average results are shown as "Recommended" in the tables.

Drag and Ballistic Coefficients

The drag coefficient was taken as 2.86, the mean of estimates received from J. Heller (LeRC) and P. Jaffe (JPL). The ballistic coefficient is:

mass/(drag coefficient x projected area)

Model Atmosphere

Taken from Kwok (ref. 1), for 2-sigma high solar activity.

Orbital Lifetime

Based on tables of D. German (ref. 2), giving altitude vs. (orbital lifetime/ballistic coefficient). German utilized Kwok's atmospheric model. For altitudes between those given in German's table, I fitted a quadratic spline to the logarithm of the altitude. Since German's tables cover altitudes of 290 km and higher, values in Table 2 for lower altitudes utilize extrapolation.

References

1) J. H. Kwok, JPL, "Drag effect on lifetime of high altitude spacecraft", IOM 312/83.2 to R. A. Wallace, 7 April 1983.

2) D. German, Science Applications, Inc., April 1986.

LDJ:tk

cc: SDAT Member & Info Lists

Table 1. Circular orbit altitude required for 1-year orbital lifetime

			Angle of Attack* 6o, spinning		Angle of Attack 90o, spinning		Angle of Attack 90o, face-on		Recommended Altitude Requirement km
	Mass, kg	Overall shape	Projected Area,** m2	Required Altitude, km	Projected Area, m2	Required Altitude, km	Projected Area, m2	Required Altitude, km	
1 antenna quad	8,820	Rod	6.8	360	32	450	32	450	390
1 antenna quad with interconnect	10,570	Rod	11	370	34	440	34	440	390
2 antenna quads	17,640	Rod	13	360	55	440	58	440	390
2 antenna quads with interconnects	21,140	Rod	13	350	57	430	60	430	380

* Angle between orbital velocity vector and plane of maximum projected area.
** Area projected normal to velocity vector

Table 2. Lifetime vs. orbit altitude for two SBR antenna quadrants.

Altitude, km	Orbital life, days				
	Angle of attack 6o		Angle of attack 90o		Recommended value
	without interconnect	with interconnect	without interconnect	with interconnect	
200	10	12	2.4	2.7	5.2
225	19	22	4.5	5.0	10
250	34	42	8.0	9.6	18
275	61	73	14	17	32
300	110	130	25	29	56
325	180	220	43	50	96
350	310	370	73	84	160
375	500	600	120	140	260
400	810	970	190	220	420
425	1300	1500	300	350	660
450	1900	2300	460	530	1000
475	2900	3500	680	790	1500
500	4300	5100	1000	1200	2200

JPL

Jet Propulsion Laboratory
California Institute of Technology

4800 Oak Grove Drive
Pasadena. California 91109

MEMORANDUM

SP-100 Project Office,
SP-100 Program DOD/DOE/NASA

Ref. No.: 313.1-86-036

For: ☐ Information ☐ Action ☐ Planning ☐ Procedure

Date: August 4, 1986

Subject: Minimal Altitude and Orbital Decay Rate During SBR Assembly

To: SDAT

From: L. Jaffe

Ref. a: IOM 313.1-86-16, "Minimal Altitude for SBR Assembly", Jaffe to SDAT, 6/23/86

Ref. b: IOM 313.1-86-030FTHL, "Orbital Decay Rate During SBR Assembly", Jaffe to SDAT, 7/16/86

SUMMARY

An altitude of 400 km or slightly lower should be adequate for assembly of the SBR spacecraft.

BASIS

Refs. (a) and (b) gave data concerning the altitude needed to provide 1-year orbital life for each of the major SBR and upper stage elements and the orbital decay rate for these elements. This memo provides corresponding data for the whole SBR spacecraft (excluding propulsion), assembled but not deployed.

Using a weighted average projected area for drag, to provide some margin for uncertainty in the spacecraft attitude, as discussed Ref. (b), the altitude required for 1 year orbital lifetime is 400 km. If the spacecraft is placed initially at 400 km, 6 months later its altitude will have fallen to 360 km. Results are summarized in Tables 1 and 2 and detailed in the appendix. Table 1 also gives results for some partial assemblies.

The element whose orbit decays most quickly is the first element left in orbit: a single radar antenna quadrant, with its associated structure (Table 1). This sets the initial altitude needed for assembly at about 410 km. The altitude of the elements left in orbit will decrease between Shuttle sorties; successive sorties, each bringing up an element to assemble, will be to successively lower altitudes.

NOTES

Recent information (being documented separately) indicates that the spacecraft can be transferred from assembly to operational orbit by a Centaur off-loaded to fit within Shuttle cargo mass constraints. (This of course ignores the recent NASA decision not to carry Centaur in Shuttle, which came too late to be considered in our study.) The Centaur would not have to be left in orbit between Shuttle sorties. The last assemblage that would be stored in orbit between Shuttle sorties is the SBR spacecraft, assembled but not deployed.

The altitude required for a given orbital lifetime of the assembled spacecraft is lower than the corresponding altitude for the Shuttle sortie cargo consisting of the SRPS with the mission module, signal processing, radar central power conditioning, and associated structural interconnects (Ref. a). The latter package would not be parked in orbit alone, but rather assembled with the previously parked antenna quadrants and structure to form the complete spacecraft.

The assumptions and data sources used in the calculations are given in Refs. (a) and (b).

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Table 1.
Circular orbit altitude required for 1-year orbital lifetime

	Mass,	Overall	Angle of Attack* 6°, spinning		Angle of Attack 90°, spinning		Suggested Values, using Weighted Mean Projected Area		
			Projected Area, **	Required Altitude for 1 year life,	Projected Area,	Required Altitude for 1 year life,	Projected Area,	Required Altitude for 1 year life,	Altitude after 0.5 year,
	kg	shape	m^2	km	m^2	km	m^2	km	km
1 antenna quad with structure	10,570	Rod	11	370	34	440	19	410	370
2 antenna quads with structure	21,140	Rod	13	350	57	430	28	390	350
Assembled spacecraft, not deployed	52,900	Rod	46	370	157	430	84	400	360

* Angle between orbital velocity vector and plane of maximum projected area.
** Area projected normal to velocity vector

Table 2. Summary of altitude decrease vs. time for SBR spacecraft, assembled but not deployed.
(weighted average projected area)

Initial altitude,	Altitude after time shown	
km	km	
	2 mos.	6 mos.
300	0	0
350	322	0
400	389	363
450	446	436
500	498	494
550	549	547
600	599	598

APPENDIX

DETAILED RESULTS

SBR spacecraft, assembled but not deployed. 6 deg angle of attack.

Mass= 52900 kg   Area= 46 m^2
For drag coefficient= 2.86 , ballistic coefficient= 402.0979 kg/m^2
2 sigma high solar activity

Altitude,	Orbital life,		Orbital decay rate		Altitude after	
					0.167 year	0.500 year
km	days	years	km/day	km/year	km	km
200.000	8.401	0.023	4.665	1703.948	0.000	0.000
210.000	10.824	0.030	3.672	1341.278	0.000	0.000
220.000	13.895	0.038	2.902	1059.776	0.000	0.000
230.000	17.775	0.049	2.301	840.514	0.000	0.000
240.000	22.658	0.062	1.832	669.135	0.000	0.000
250.000	28.778	0.079	1.464	534.716	0.000	0.000
260.000	36.423	0.100	1.174	428.920	0.000	0.000
270.000	45.935	0.126	0.946	345.364	0.000	0.000
280.000	57.724	0.158	0.764	279.143	0.000	0.000
290.000	72.282	0.198	0.620	226.479	223.004	0.000
300.000	90.189	0.247	0.505	184.454	254.833	0.000
310.000	112.133	0.307	0.413	150.802	276.173	0.000
320.000	138.921	0.380	0.339	123.763	293.463	0.000
330.000	171.497	0.470	0.279	101.963	308.615	0.000
340.000	210.959	0.578	0.231	84.326	322.468	253.749
350.000	258.579	0.708	0.192	70.010	335.464	292.381
360.000	315.823	0.865	0.160	58.350	347.863	317.023
370.000	384.368	1.052	0.134	48.821	359.828	336.464
380.000	466.129	1.276	0.112	41.008	371.466	353.176
390.000	563.272	1.542	0.095	34.580	382.850	368.216
400.000	678.242	1.857	0.080	29.274	394.031	382.135
410.000	813.776	2.228	0.068	24.880	405.042	395.255
420.000	972.925	2.664	0.058	21.229	415.907	407.776
430.000	1159.066	3.173	0.050	18.186	426.643	419.833
440.000	1375.913	3.767	0.043	15.641	437.259	431.517
450.000	1623.265	4.444	0.038	13.793	447.636	442.713
460.000	1909.420	5.228	0.032	11.838	457.950	453.761
470.000	2242.572	6.140	0.028	10.176	468.235	464.661
480.000	2629.810	7.200	0.024	8.762	478.494	475.436
490.000	3075.634	8.421	0.021	7.751	488.678	486.031
500.000	3583.021	9.810	0.018	6.699	498.776	496.486
510.000	4169.766	11.416	0.016	5.796	508.867	506.892
520.000	4847.543	13.272	0.014	5.020	518.955	517.248
530.000	5629.622	15.413	0.012	4.353	529.040	527.563
540.000	6531.071	17.881	0.010	3.779	539.123	537.843
550.000	7568.979	20.723	0.009	3.283	549.204	548.093
560.000	8762.692	23.991	0.008	2.856	559.284	558.319
570.000	10134.110	27.746	0.007	2.488	569.362	568.523
580.000	11707.960	32.055	0.006	2.169	579.439	578.708
590.000	13512.150	36.994	0.005	1.893	589.514	588.876
600.000	15578.130	42.651	0.005	1.654	599.585	599.028

SPR Spacecraft, assembled but not deployed. 90 deg angle of attack.

Mass= 52900 kg   Area= 157 m^2
For drag coefficient= 2.86 , ballistic coefficient= 117.8121 kg/m^2
2 sigma high solar activity

Altitude,	Orbital life,		Orbital decay rate		Altitude after	
					0.167 year	0.500 year
km	days	years	km/day	km/year	km	km
200.000	2.461	0.007	15.922	5815.648	0.000	0.000
210.000	3.171	0.009	12.533	4577.842	0.000	0.000
220.000	4.071	0.011	9.903	3617.064	0.000	0.000
230.000	5.208	0.014	7.854	2868.712	0.000	0.000
240.000	6.639	0.018	6.253	2283.788	0.000	0.000
250.000	8.432	0.023	4.997	1825.010	0.000	0.000
260.000	10.672	0.029	4.008	1463.924	0.000	0.000
270.000	13.459	0.037	3.227	1178.741	0.000	0.000
280.000	16.913	0.046	2.608	952.726	0.000	0.000
290.000	21.178	0.058	2.116	772.984	0.000	0.000
300.000	26.425	0.072	1.724	629.549	0.000	0.000
310.000	32.854	0.090	1.409	514.693	0.000	0.000
320.000	40.703	0.111	1.156	422.407	0.000	0.000
330.000	50.247	0.138	0.953	348.002	0.000	0.000
340.000	61.810	0.169	0.788	287.810	186.026	0.000
350.000	75.762	0.207	0.654	238.949	275.593	0.000
360.000	92.534	0.253	0.545	199.152	307.451	0.000
370.000	112.617	0.308	0.456	166.629	330.006	0.000
380.000	136.573	0.374	0.383	139.962	348.465	0.000
390.000	165.035	0.452	0.323	118.022	364.621	0.000
400.000	198.721	0.544	0.274	99.913	379.309	279.038
410.000	238.431	0.653	0.232	84.915	392.986	333.709
420.000	285.061	0.780	0.198	72.455	405.926	363.822
430.000	339.599	0.930	0.170	62.068	418.306	386.323
440.000	403.134	1.104	0.146	53.383	430.244	405.029
450.000	475.606	1.302	0.129	47.078	441.640	421.207
460.000	559.448	1.532	0.111	40.403	452.846	435.943
470.000	657.059	1.799	0.095	34.731	463.885	449.823
480.000	770.518	2.110	0.082	29.904	474.776	463.028
490.000	901.141	2.467	0.072	26.454	485.467	475.569
500.000	1049.802	2.874	0.063	22.863	495.995	487.490
510.000	1221.715	3.345	0.054	19.782	506.470	499.226
520.000	1420.299	3.889	0.047	17.134	516.884	510.692
530.000	1649.443	4.516	0.041	14.857	527.248	521.937
540.000	1913.562	5.239	0.035	12.896	537.571	533.002
550.000	2217.663	6.072	0.031	11.207	547.857	543.918
560.000	2567.413	7.029	0.027	9.749	558.114	554.710
570.000	2969.229	8.129	0.023	8.490	568.345	565.397
580.000	3430.358	9.392	0.020	7.402	578.553	575.996
590.000	3958.974	10.839	0.018	6.461	588.741	586.518
600.000	4564.294	12.496	0.015	5.645	598.911	596.976

SBR spacecraft, assembled but not deployed. Weighted average projected area.

Mass= 52900 kg   Area= 84 m^2
For drag coefficient= 2.86 , ballistic coefficient= 220.1965 kg/m^2
2 sigma high solar activity

Altitude, km	Orbital life, days	Orbital life, years	Orbital decay rate km/day	Orbital decay rate km/year	Altitude after 0.167 year km	Altitude after 0.500 year km
200.000	4.601	0.013	8.519	3111.557	0.000	0.000
210.000	5.927	0.016	6.706	2449.291	0.000	0.000
220.000	7.609	0.021	5.298	1935.244	0.000	0.000
230.000	9.734	0.027	4.202	1534.852	0.000	0.000
240.000	12.408	0.034	3.345	1221.900	0.000	0.000
250.000	15.760	0.043	2.673	976.439	0.000	0.000
260.000	19.946	0.055	2.144	783.246	0.000	0.000
270.000	25.155	0.069	1.727	630.664	0.000	0.000
280.000	31.611	0.087	1.396	509.739	0.000	0.000
290.000	39.583	0.108	1.132	413.571	0.000	0.000
300.000	49.389	0.135	0.922	336.829	0.000	0.000
310.000	61.406	0.168	0.754	275.377	162.261	0.000
320.000	76.076	0.208	0.619	226.001	252.718	0.000
330.000	93.915	0.257	0.510	186.192	282.683	0.000
340.000	115.525	0.316	0.422	153.987	304.021	0.000
350.000	141.603	0.388	0.350	127.845	321.606	0.000
360.000	172.951	0.474	0.292	106.553	337.103	0.000
370.000	210.487	0.576	0.244	89.152	351.284	275.973
380.000	255.261	0.699	0.205	74.884	364.571	316.831
390.000	308.459	0.845	0.173	63.146	377.216	342.774
400.000	371.418	1.017	0.146	53.456	389.380	363.102
410.000	445.639	1.220	0.124	45.432	401.172	380.456
420.000	532.793	1.459	0.106	38.766	412.664	395.972
430.000	634.727	1.738	0.091	33.208	423.908	410.243
440.000	753.476	2.063	0.078	28.561	434.940	423.616
450.000	888.931	2.434	0.069	25.188	445.640	436.112
460.000	1045.635	2.863	0.059	21.617	456.247	448.167
470.000	1228.075	3.362	0.051	18.582	466.777	459.945
480.000	1440.134	3.943	0.044	15.999	477.244	471.441
490.000	1684.276	4.611	0.039	14.154	487.589	482.627
500.000	1962.131	5.372	0.033	12.233	497.836	493.537
510.000	2283.443	6.252	0.029	10.584	508.056	504.362
520.000	2654.607	7.268	0.025	9.167	518.253	515.072
530.000	3082.888	8.440	0.022	7.949	528.432	525.687
540.000	3576.539	9.792	0.019	6.900	538.595	536.222
550.000	4144.917	11.348	0.016	5.996	548.746	546.691
560.000	4798.617	13.138	0.014	5.216	558.886	557.103
570.000	5549.631	15.194	0.012	4.543	569.016	567.468
580.000	6411.502	17.554	0.011	3.960	579.137	577.791
590.000	7399.511	20.259	0.009	3.457	589.250	588.077
600.000	8530.882	23.356	0.008	3.020	599.355	598.332

JET PROPULSION LABORATORY　　　　　INTEROFFICE MEMORANDUM

16 SEPTEMBER 1986

TO: C. Bell
J. M. Boudreau

FROM: L. Jaffe

SUBJECT: Orbital Lifetime of Assembled SBR Spacecraft (further revised)

Refs.:

a. IOM 313.11-86-017FTHL, L.Jaffe to C.Bell & J.M.Boudreau, this subject, 23 June 1986

b. IOM 343-86-1129, T.Kia to L.Jaffe, "SP-100 orientation using passive gravity gradient stabilization," 6 August 1986

c. IOM 343-86-1170, T.Kia to L.Jaffe, "SP-100 orientation using passive gravity gradient stabilization, revisited," 12 August 1986

d. IOM 343-86-169, J.Spanos, "SP-100 attitude control, system design, and integration review," 12 February 1986

e. J.M. Boudreau, "Safety of Start-up in (High) Shuttle Orbit", Los Alamos Natl. Lab, in prep.

f. J. Sercel, "Propulsion Subsystem," appendix to R.M. Jones, "SP-100 SBR Study. Final Review", July 16, 1985.

Reference (a) gave the orbital lifetime of the assembled SBR spacecraft if electric propulsion should cease during the transfer from assembly orbit to operational orbit. It utilized information on SBR attitude given in reference (b), and covered initial altitudes of 420-500 km.

The spacecraft attitudes given in reference (b) have since been updated by reference (c). This memo updates the orbital lifetimes accordingly. Results are given in the attached Table 1 and Figure 1. Taken into account is the effect of the separation distance between reactor and mission module upon the stable attitude (reference c) and hence upon the orbital lifetime. Because we are now considering assembly orbits lower than 420 km for the SBR, trajectories have been added for initial altitudes down to 380 km.

The table includes ion as well as arcjet propulsion but our current scenarios do not utilize ion propulsion for the SBR mission.

Comparing the results with your curves for desired time for decay of radioactivity, it appears that there is a problem only during the first weeks of operation, as you previously pointed out. There appears to be significant advantage, with respect to decay time provided, in increasing the separation distance to 40 meters and increasing the assembly altitude to 500 km.

It is possible to increase the altitude attained at any time during the first weeks of operation by lowering the specific impulse of the propulsion system (reference f). With arcjets the specific impulse might be lowered from the 1000 lbf-s/lbm, used in the table, to 500 lbf-s/lbm, for example. This would in principle permit attaining a given altitude in half the time, at the cost of added propellant mass. The added propellant mass is not of great concern, since the propulsion system could still be launched by 1 Shuttle flight. However, the engine efficiency would drop, neces-sitating a calculation I have not yet made. Also, looking at the figure, it appears that even halving the time to reach a given altitude, and hence a given orbital life-time, will not in itself avoid the problem during the first few weeks: the curves will still cross.

To aid in estimating how high the start-up altitude would have to be to provide decay times above your curves throughout the propulsion period, I have added to the table information on the orbital lifetime for various given altitudes and plotted these results (Figure 2). Though your curves of decay time vs. run time, shown in Figure 1 (reference e) extend down only to run times of 7 days, they suggest that decay times of 10 to 100 years should be acceptable for very short operating times. Figure 2 indicates that such orbital lifetimes correspond to SBR altitude of 850 +/- 50 km with a separation distance of 25 meters and 660 +/- 40 km with a separation distance of 40 m.

Data sources and assumptions used in the orbital lifetime calculation, as well as further comments, are attached.

cc: SDAT member and info lists

TABLE 1

ORBITAL LIFETIME FOR DEPLOYED SBR SPACECRAFT
IF PROPULSION CEASES DURING TRANSFER TO OPERATIONAL ORBIT

Propulsion	Burn time days	Initial orbit km	Delta-V m/s	Fractional delta-V (fractional transfer time)*	Altitude km	Inclination deg	Orbital lifetime, years Near Edge-on Solar activity High	Nominal	Face-on Solar activity High	Nominal	Suggested value at separation** = 25 m	40 m
None	0	380	0	0.000	380	57.0	2.8E-1	(4.5E-1)	2.8E-2	(4.5E-2)	2.8E-2	5.1E-2
	0	400	0	0.000	400	57.0	4.0E-1	(6.9E-1)	4.0E-2	(6.9E-2)	4.0E-2	8.3E-2
	0	420	0	0.000	420	57.0	5.8E-1	(1.0E0)	5.8E-2	(1.0E-1)	5.8E-2	1.4E-1
	0	450	0	0.000	450	57.0	9.6E-1	(1.9E0)	9.6E-2	(1.9E-1)	9.6E-2	2.8E-1
	0	480	0	0.000	480	57.0	1.6E0	(3.2E0)	1.6E-1	(3.2E-1)	1.6E-1	5.8E-1
	0	500	0	0.000	500	57.0	2.1E0	(4.6E0)	2.1E-1	(4.6E-1)	2.1E-1	8.7E-1
	0	520	0	0.000	520	-	2.9E0	(6.6E0)	2.9E-1	(6.6E-1)	2.9E-1	1.4E0
	0	540	0	0.000	540	-	3.9E0	(9.3E0)	3.9E-1	(9.3E-1)	3.9E-1	2.1E0
	0	560	0	0.000	560	-	5.2E0	(1.3E1)	5.2E-1	(1.3E0)	5.2E-1	3.1E0
	0	580	0	0.000	580	-	6.9E0	(1.8E1)	6.9E-1	(1.8E0)	6.9E-1	4.5E0
	0	600	0	0.000	600	-	9.2E0	(2.5E1)	9.2E-1	(2.5E0)	9.2E-1	6.5E0
	0	620	0	0.000	620	-	1.2E1	(3.4E1)	1.2E0	(3.4E0)	1.2E0	9.1E0
	0	640	0	0.000	640	-	1.6E1	(4.6E1)	1.6E0	(4.6E0)	1.6E0	1.3E1
	0	660	0	0.000	660	-	2.1E1	(6.2E1)	2.1E0	(6.2E0)	2.1E0	1.7E1
	0	680	0	0.000	680	-	(2.8E1)	8.3E1	2.8E0	(8.3E0)	2.8E0	7.9E1
	0	700	0	0.000	700	-	(3.6E1)	1.1E2	3.6E0	(1.1E1)	3.6E0	1.1E2
	0	720	0	0.000	720	-	(4.7E1)	1.5E2	4.7E0	(1.5E1)	4.7E0	1.5E2
	0	740	0	0.000	740	-	(6.1E1)	1.9E2	6.1E0	(1.9E1)	6.1E0	1.9E2
	0	760	0	0.000	760	-	(7.8E1)	2.5E2	7.8E0	(2.5E1)	7.8E0	2.5E2
	0	780	0	0.000	780	-	(1.0E2)	3.1E2	1.0E1	(3.1E1)	1.0E1	3.1E2
	0	800	0	0.000	800	-	(1.3E2)	4.0E2	1.3E1	(4.0E1)	1.3E1	4.0E2
	0	820	0	0.000	820	-	(1.6E2)	4.9E2	1.6E1	(4.9E1)	1.6E1	4.9E2
	0	840	0	0.000	840	-	(2.0E2)	6.1E2	2.0E1	(6.1E1)	2.0E1	6.1E2
	0	860	0	0.000	860	-	(2.5E2)	7.5E2	(2.5E1)	7.5E1	7.5E1	7.5E2
	0	880	0	0.000	880	-	(3.0E2)	9.1E2	(3.0E1)	9.1E1	9.1E1	9.1E2
	0	900	0	0.000	900	-	(3.7E2)	1.1E3	(3.7E1)	1.1E2	1.1E2	1.1E3
	0	950	0	0.000	950	-	(6.0E2)	1.7E3	(6.0E1)	1.7E2	1.7E2	1.7E3
	0	1000	0	0.000	1000	-	(9.2E2)	2.4E3	(9.2E1)	2.4E2	2.4E2	2.4E3
	0	1088	0	0.000	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3
Arc	7	380	250	0.245	579	58.0	6.8E0	(1.8E1)	6.8E-1	(1.8E0)	6.8E-1	4.5E0
	7	400	250	0.250	594	58.0	8.5E0	(2.3E1)	8.5E-1	(2.3E0)	8.5E-1	6.0E0
	7	420	250	0.255	610	58.0	1.1E1	(2.9E1)	1.1E0	(2.9E0)	1.1E0	8.0E0
	7	450	250	0.265	636	58.0	1.5E1	(4.3E1)	1.5E0	(4.3E0)	1.5E0	1.2E1
	7	500	250	0.285	684	58.0	(2.9E1)	8.8E1	2.9E0	(8.8E0)	2.9E0	8.4E1
	28.0	380	998	0.981	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3
	27.5	400	978	0.981	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3
	26.8	420	956	0.981	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3
	25.9	450	921	0.980	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3
	24.1	500	856	0.978	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3
Ion	7	380	117	0.115	482	57.4	1.6E0	(3.4E0)	1.6E-1	(3.4E-1)	1.6E-1	5.9E-1
	7	400	117	0.117	497	57.4	2.0E0	(4.4E0)	2.0E-1	(4.4E-1)	2.0E-1	8.3E-1
	7	420	117	0.120	513	57.4	2.6E0	(5.9E0)	2.6E-1	(5.9E-1)	2.6E-1	1.2E0
	7	450	117	0.124	539	57.4	3.8E0	(9.2E0)	3.8E-1	(9.2E-1)	3.8E-1	2.0E0
	7	500	117	0.133	585	57.4	7.5E0	(2.0E1)	7.5E-1	(2.0E0)	7.5E-1	4.9E0
	30	380	500	0.492	751	58.9	(7.0E1)	2.2E2	7.0E0	(2.2E1)	7.0E0	2.2E2
	30	400	500	0.501	765	58.9	(8.3E1)	2.6E2	8.3E0	(2.6E1)	8.3E0	2.6E2
	30	420	500	0.513	782	58.9	(1.0E2)	3.2E2	1.0E1	(3.2E1)	1.0E1	3.2E2
	30	450	500	0.532	809	58.9	(1.4E2)	4.4E2	1.4E1	(4.4E1)	1.4E1	4.4E2
	30	500	500	0.570	857	58.9	(2.4E2)	7.3E2	(2.4E1)	7.3E1	7.3E1	7.3E2
	58.8	400	978	0.981	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3
	57.4	420	956	0.981	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3
	55.3	450	921	0.980	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3
	51.4	500	856	0.978	1088	61.0	(1.8E3)	4.2E3	(1.8E2)	4.2E2	4.2E2	4.2E3

* Fraction of delta-V (and time) needed to reach 1100 km.
** Separation: reactor to mission module.

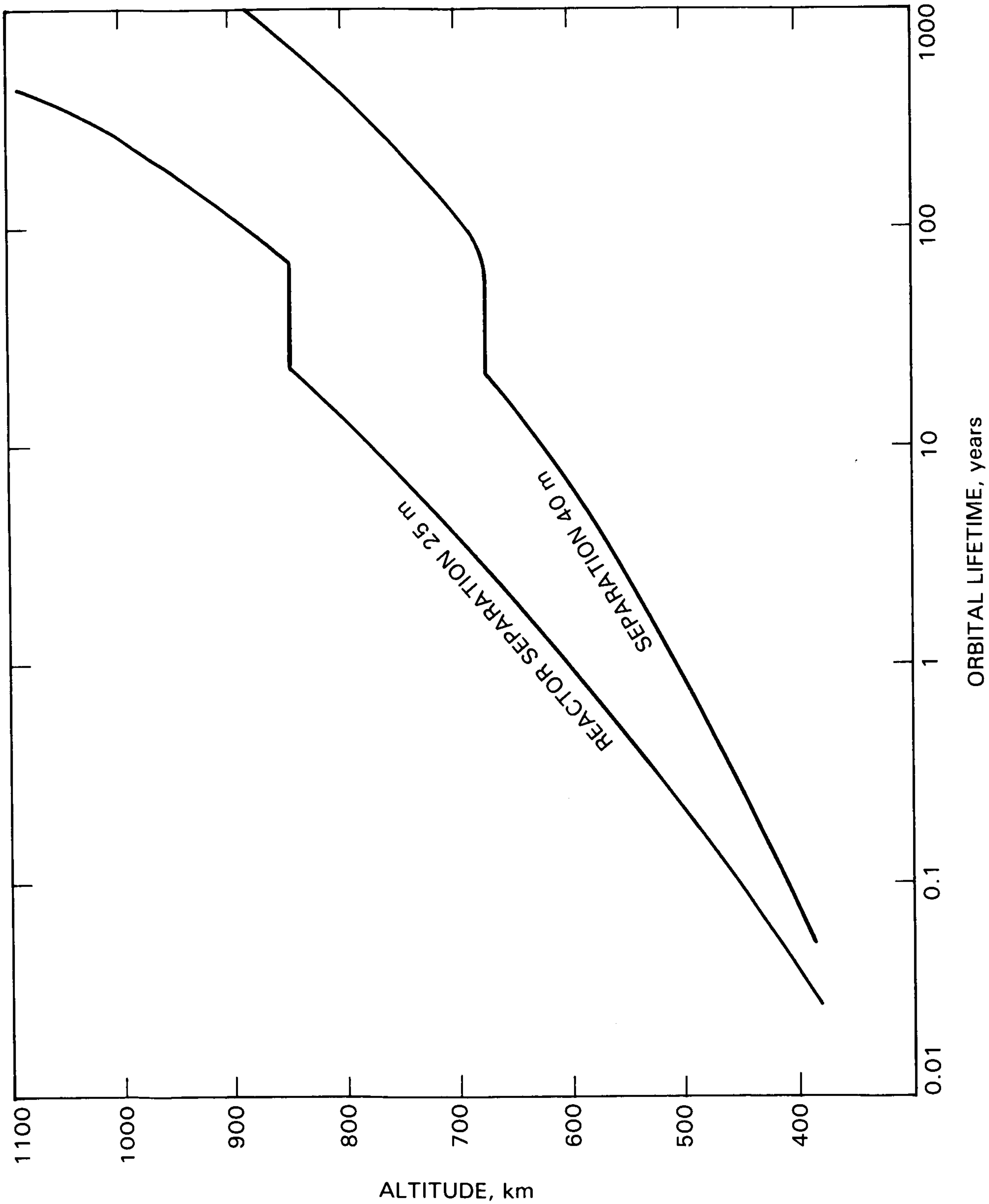


Figure 2. Orbital Lifetime of Deployed SBR vs Altitude

DATA SOURCES, ASSUMPTIONS, AND COMMENTS ON ORBITAL DECAY CALCULATIONS

SBR SPACECRAFT MASS

Taken as 55,000 kg.

SPACECRAFT ATTITUDE AND PROJECTED AREA

Orbital lifetimes were calculated for two cases:

Case 1. Face-on. 90 deg angle of attack.

SBR radar antenna (32 x 64 m) and main radiator (160 m^2) perpendicular to orbital velocity vector: projected area = 32 x 64 + 160 = 2208 m^2. This is slightly over-conservative since with the current configurations antenna and radiator cannot simultaneously be perpendicular to the velocity vector.

Case 0. Near edge-on. 6 deg average angle of attack.

Projected area taken as 0.1 * that of the face-on case. ($\sin 6^0 \sim 0.1$)

The attitude the spacecraft would assume if active attitude were off was estimated. Reference (c) and conversation with T. Kia indicate that at high altitudes, where the greatest external torque is due to gravity gradient, our 300 kWe roll-out flat plate configuration will tend to orient with the long axis of the radar antenna vertical and the boom axis along the orbital velocity vector. Conversation with L. Jaffe of JPL indicated that at low altitudes, where the greatest external torque is due to atmospheric drag, the boom axis will again tend to align with the orbital velocity vector. The spacecraft will oscillate about these stable orientations but to be conservative I suggest assuming that the boom axis is parallel to the orbital velocity vector (radar antenna face-on).

Kia points out (reference c) that by lengthening the boom to increase the reactor-to-antenna distance to 40 meters, from our strawman 25 meters, the moments of inertia can be changed so that the stable configuration under gravity gradient torque becomes boom axis vertical, long axis of antenna along the velocity vector. This spacecraft will then oscillate about an edge-on attitude of the antenna and its orbital lifetime at the higher altitudes will increase considerably. We have not examined the effects on the spacecraft of increasing the separation to 40 meters, but I think they would be acceptable.

It should be noted that Kia's calculations are for the roll-out flat plate 300 kWe configuration. The separation distance at which the axis of minimum moment of inertia changes depends on the configuration. For the 300 kWe daisy, it will be shorter than for the flat plate.

In Table 1, I suggest lifetimes for both 25- and 40-m booms. The values for 25 meters are for face-on. The values for 40 meters are based on averages of the face-on and near-edge-on projected areas, weighted by estimated relative magnitudes of the gravity gradient and aerodynamic torques at each altitude. These torques were taken from reference (d); the aerodynamic torques were scaled by atmospheric density.

SBR DRAG COEFFICIENT

Taken as 2.86, the mean of estimates received from LeRC (J. Heller) and JPL (L. Jaffe) for a flat plate in the free molecular flow regime.

SBR PROPULSION CHARACTERISTICS

Two cases:
NH_3 arc jets
Xe ion thrusters

Our selected scenario for electric propulsion uses NH_3 arc jets; ion thrusters are shown for comparison only. Propulsion system performance and mass characteristics were taken from Jaffe and Fujita (ref. 1). The specific impulse values chosen were those designated as "low to medium risk for the time period 1995-2000, namely 1000 lbf-s/lbm for arc jets and 3000 for ion thrusters.

INITIAL ORBIT

Five cases: altitudes of 380, 400, 420, 450, and 450 km, all circular, 57° inclination.

These are taken to be representative of possible orbits for SBR spacecraft assembly. An altitude of about 400 km is the minimum needed to provide 1-year orbital life for components of the spacecraft parked in orbit during assembly, during the multiple Shuttle sorties needed (L. Jaffe, ref. 2). The orbit might decay to about 380 km during the assembly process. Other altitudes were run to explore sensistivity. 57 deg inclination is the maximum allowable for a Cape Canaveral launch. (Vandenberg launchs give poorer performance.)

OPERATIONAL ORBIT

Taken as 1088 km circular at 61° inclination. Lincoln Laboratories study gave 1100 km as SBR mission requirement. Strawman orbital altitude in current SP-100 system study is 1088 km, selected as being more stable under perturbing forces than the slightly higher 1100 km. (Uphoff, ref 3).

ORBITAL TRANSFER TIME TO 1100 km

Total transfer times from initial circular orbits at 420 and 450 km to circular orbit at 1100 km were calculated by W. Gray (ref. 4) under the above assumptions. Times from other altitudes were derived by extrapolation.

ALTITUDE AND INCLINATION VS. DELTA-V

Altitude and inclination during transfer from initial 500 km orbit to 1100 km circular orbit were tabulated by R. Beatty (ref. 5) as functions of propulsion velocity increment (delta-V). Extrapolated to give delta-V from other initial altitudes (cubic spline fit).

ALTITUDE AND INCLINATION VS. BURN TIME

Joan Boudreau (Los Alamos) gave operational times for which she has calculated radioactivity build-up as 7, 30, 61 days and specified longer periods. These times were converted to fractions of the total transfer time to 1100 km. Corresponding fractional delta-V's were taken as equal to the fractional transfer times. (This approximation is reasonable since the propellant mass is small compared to the dry mass of the spacecraft). Corresponding altitudes and inclinations were then obtained by interpolation (cubic spline fit) in the tables of fractional delta-V's (= fractional transfer times) vs. altitude and inclination for each initial altitude. Inclination vs. burn time during transfer from other altitudes was taken as the same as for transfer from 500 km.

ORBITAL TRANSFER TIME TO 1088 km

Transfer times from initial circular orbits to circular orbit at 1088 km were obtained by interpolation in the tables of fractional transfer time vs. altitude. The times to reach 1088 km are shown in Table 1.

ORBITAL LIFETIMES

Orbital lifetimes were calculated for the altitudes attained after burn times of 7 and 30 days and for the initial and operational altitudes, using the assumptions stated above and tables by German (ref. 6). These tables give, as a function of altitude, the ratio of orbital lifetime to ballistic coefficient. (The ballistic coefficient is the ratio of the mass to the product of the area and the drag coefficient.) German's tables were generated from a model of Kwok (ref. 7) and form the basis for the report by German and Friedlander (ref. 8). They cover two cases,

nominal solar activity,
2-sigma high solar activity,

over the altitude range 283 to 1586 km. A quadratic spline fit to the logarithm of the lifetime was used for interpolation to the desired altitudes.

The attached table lists values of orbital lifetime for each burn time and initial orbit, covering the 2 choices of projected area (edge-on and face-on) and 2 states of solar activity (high and nominal). The appropriate choice of solar activity depends on the orbital lifetime as compared to the 11-year length of a solar cycle. For lifetimes shorter than two solar cycles, I consider high solar activity appropriate as a worst case; for longer lifetimes the atmospheric density should be averaged over the solar cycles; I take the model for nominal solar activity to be appropriate. In the table, values for the solar activity level considered more appropriate for each spacecraft attitude are shown without parentheses; those for the solar activity level considered less appropriate are shown in parentheses.

The tabulated lifetimes for edge-on attitude are 10 x those for face-on. As explained under "Spacecraft attitude and projected area", the expected attitude without active attitude control is face-on for a reactor-to-radar antenna separation distance of 25 m (our current strawman design). If the separation distance is increased to 40 m, the expected attitude would be near edge-on for high altitudes and face-on for low altitudes. The last columns of the table give corresponding lifetime values, under the heading "Suggested value". Lifetimes for 40-m separation use weighted average projected areas as explained in the section "Spacecraft attitude and projected area".

The horizontal jog in the curves of Figure 2 occurs when they cross 22 years (2 solar cycles) and the solar activity assumed for the atmospheric model changes from 2-sigma high to nominal. This is reflected by the choice of columns in Table 1, indicated by the parentheses.

REFERENCES

1) L. Jaffe and T. Fujita, JPL, "Electric propulsion characteristics," IOM 313.1.86-15 to SDAT, 12 June 1986.

2) L. Jaffe, JPL, "Minimal altitude and orbital decay rate during SBR assembly", IOM 313.1-86-036 to SDAT, August 4, 1986.

3) C. Uphoff, JPL, "Preliminary assessment of perturbations for SP100 orbit", IOM 21/85.2-955 to R. Jones, 12 April 1985.

4) W. Gray, JPL, "SBR performance for SBR and OTV applications," IOM 311.3-1631 to L. Jaffe, 25 June 1986.

5) R. Beatty, JPL, "Altitude and inclination as a function of time", IOM to L. Jaffe, 18 March 1986.

6) D. German and A. Freidlander, Science Applications International, "Nuclear safe orbit raising analysis", report SAIC-85/1911 on JPL contract 956817, November 1985.

7) J. H. Kwok, JPL, "Drag effect on lifetime of high altitude spacecraft" IOM 312/83.2-689 to R. A. Wallace, 7 April 1983.

B-8. SCENARIO USING CHEMICAL PLUS ELECTRIC PROPULSION FOR TRANSFER FROM ASSEMBLY ORBIT TO OPERATIONAL ORBIT

J. Heller
H. Bloomfield

August 1986

A Shuttle/OMV Scenario for Orbit Placement
of an SP-100 Powered Space Based Radar

A. Initial Conditions/Assumptions:

1. SRB's mission is in a circular orbit @ 61° inclination at 1080 km.

2. SRB consists of basically 4 separate quadrants of stowed flat antennae, a pair of which can be packed into the Shuttle bay and weigh no more than 21,000 kg per pair. The size and mass of the antennae will have to be reduced to meet STS cargo mass limitations.

3. At the time of IOC of the SRB the STS will be improved to have a cargo delivery capability of over 22,000 kg @ 57 ° inclination at 278 km (150 n.mi).

4. Assume OMV characteristics of the MSFC reference design:

   Fully loaded $\cong$ 5900 kg (incl 100 kg ASE)
   Fuel = 3200 kg bi prop.

5. Assume a 300 kW_e SP-100 mass of 9900 kg., including boom and power conditioning.

6. From MSFC data the present reference design OMV can raise a 54,500 kg payload to over 250 km and de-orbit empty to 200 km.

7. 700 km is the minimum acceptable altitude at which the nuclear power system can be started.

B. Proposed procedure for SBR assembly and insertion into final mission orbit:

1. Shuttle Flt. #1 delivers a pair of folded antennae quadrants to 278 km at a 57° inclination and removed from the cargo bay by the RMS arm.

2. Shuttle Flt. #2 delivers the remaining pair of antennae quadrants to 278 km and the pairs of quadrants are joined by the RMS and minimal EVA. To be determined are small orbit maintenance thruster systems to account for long periods between Shuttle launches.

3. Shuttle Flt. #3 delivers the 300 kW_e SP-100 reactor power system (SRPS), any auxillary SBR equipment and two fully-loaded OMV's to the 278 km staging orbit @ 57° inclination. (Total mass ~ 21,700 kg).

4. The SBR antenna is fully deployed and the balance of SBR equipment assembled and checked out. The SP-100 is then fully deployed and attached to the SBR planar antennae array. One fully loaded OMV is then employed to raise the orbit of the entire structure of about 58,600 kg to 475 km orbit. It appears cost-effective to upgrade the performance of special OMV's in order to greatly reduce the number of Shuttle flights from 5 or 6 to 3.

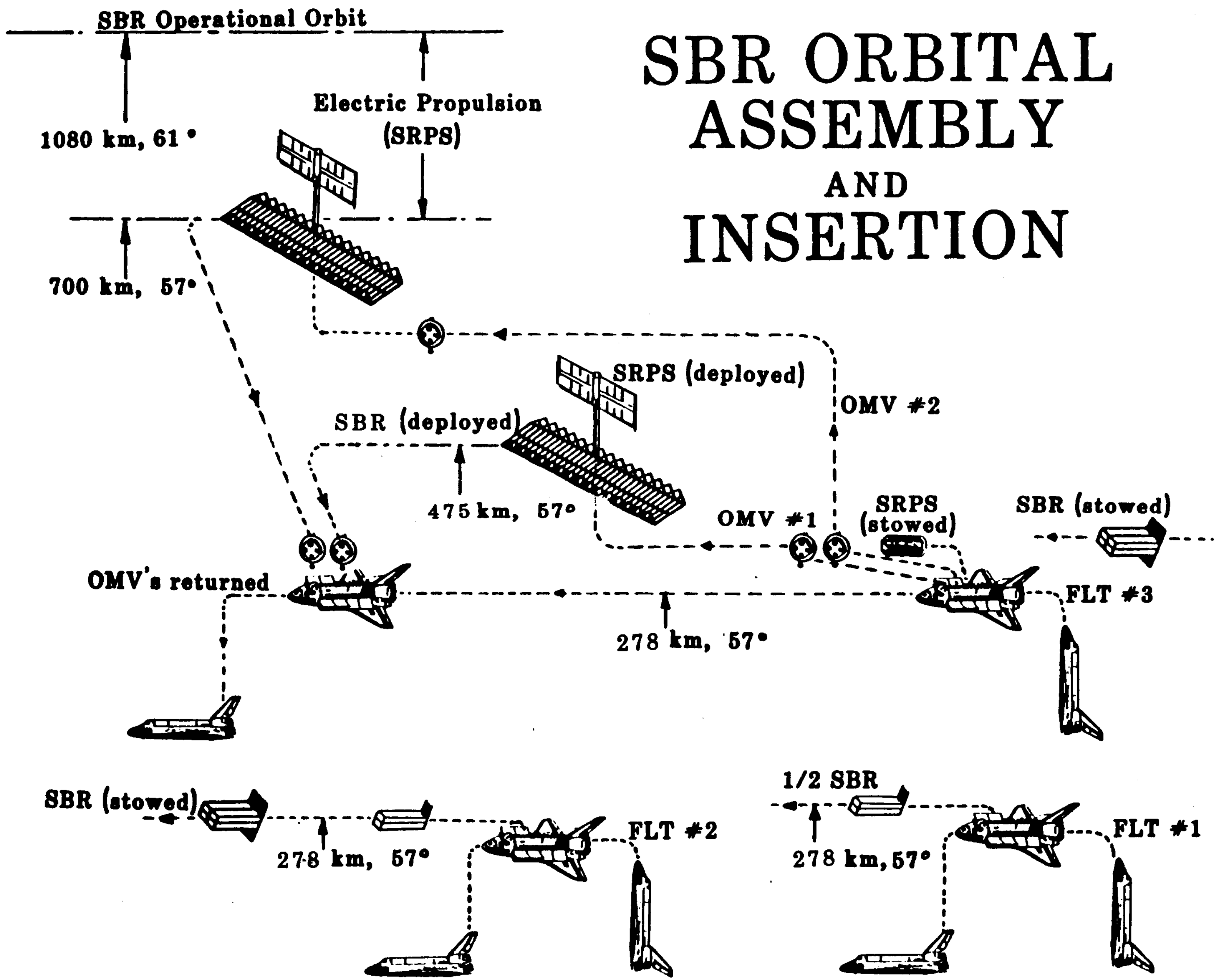
5. At 475 km orbit the first OMV returns to the Shuttle empty, and the second full OMV raises the package to the "minimum reactor start" altitude of 700 km. The second OMV returns empty to the Shuttle.

6. At 700 km the reactor power system is fully checked out and started either from controls in the Shuttle or a ground station. The SBR, now with high power, can also be checked out. The SBR is then placed in final orbit @ 61^{o} and 1080 km, employing electric thrusters to attain its operational orbit. The mass of a minimum-size electric thruster system will have to be factored into the total scenario to maintain the 3 shuttle SBR launch concept.

SBR ORBITAL ASSEMBLY & INSERTION

- **FLT #1 1/2 SBRto 278 km, 57°; 23,000 kg,max.**
- **FLT #2 Bal. SBR to 278 km**
- **SBR halves joined by RMS and EVA**
- **FLT #3 SRPS,2 OMV's, & balance of SBR, to 278 km, 57°; 23,000 kg,max.**
- **Deploy SBR:, attach/ deploy SRPS,engage OMV, checkout, boost to 475 km,57° with #1 OMV**
- **At 475 km, exchange OMV's and boost package to 700 km. OMV's returned via a shuttle.**
- **At 700 km start SRPS, use electric propulsion to boost to 1080 km, 61° operational orbit.**

SBR ORBITAL ASSEMBLY AND INSERTION



B-9. TEMPERATURE CONTROL DURING EXTENDED ORBITAL STORAGE

P. Bhandari
C. Cagle
J. Stallkamp

June-July 1986

JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM

3546-TSE-86-121

June 18, 1986

TO: Len Jaffe

FROM: Pradeep Bhandari

SUBJECT: Effect of Solar Absorptance of the MLI Blanket on the Power Conditioning Module Temperatures in Storage Orbit

INTRODUCTION

This memo is an extension of Cathy Cagle's memo (IOM #3546-SP100-86-003, "SP100 Space-Based-Radar in Orbital Storage"). A copy of the memo is attached for reference.

The results of further thermal analyses of the power conditioning module (PCC) and its radiators in an extended storage orbit (450 km) are presented here.

SOLAR ABSORPTANCE OF MLI BLANKET

In the storage orbit, Cathy's calculation showed that the blanketed PCC's temperature would lie in the range of -70 to -30°C, if no heater is available. It was also found that a heater power of 100 to 160 watts would be required to ensure that the PCC temperature is controllable at 0 to 10°C. The solar absorptance of the MLI blanket assumed in her analysis was 0.2 (the IR absorptance assumed is 0.8).

In order to eliminate or reduce the heater power requirements, this thermal analyses was performed to study the effect of the MLI solar absorptance on the PCC temperatures. All the previous assumptions and calculation procedures were retained excepting the value of the solar absorptance.

RESULTS

Parametric computer model runs were made. The solar absorptance was varied from 0.2 to 0.8. The criteria for the choice of the right absorptance were the hottest orbit temperatures at zero heater power. The maximum absorptance at which this temperature was reasonable (~ 20°C) was found to be 0.7. At this value of absorptance, the coldest orbit temperature is calculated to be equal to -24°C (zero heater power).

According to John Stallkamp, a safe temperature range for the non-operational PCC (in storage) is -50 to +100°C. Hence, the above configuration, with an MLI solar absorptance of 0.7, would be well within this allowed range.

Figure 1 shows pictorially the expected range of the PCC temperatures at various heater power levels, assuming an absorptance of 0.7.

In Cathy's calculation, it was assumed that for any orbit (hot or cold), because of the large thermal mass of the system (PCC) and relatively low heat transfer rates, the heat loads (or losses) could be averaged over the whole orbit. In other words, the PCC temperature would not vary too much within each orbit, due to varying heat loads (or losses). A transient analyses was done to verify this. The PCC mass was provided by John Stallkamp. A 50 kg mass was used in the analysis. The temperature fluctuation within an orbit turns out to be on the order of 3-4^{o}C, which is quite small. This verifies the validity of that assumption.

CONCLUSIONS

A solar absorptance of 0.7 for the MLI blanket will suffice to keep the PCC temperatures (in the storage orbit) in the range of -24 to +22^{o}c. No heater power is required to achieve this.

A transient thermal analysis verifies the assumption of using average heat loads within each orbit.

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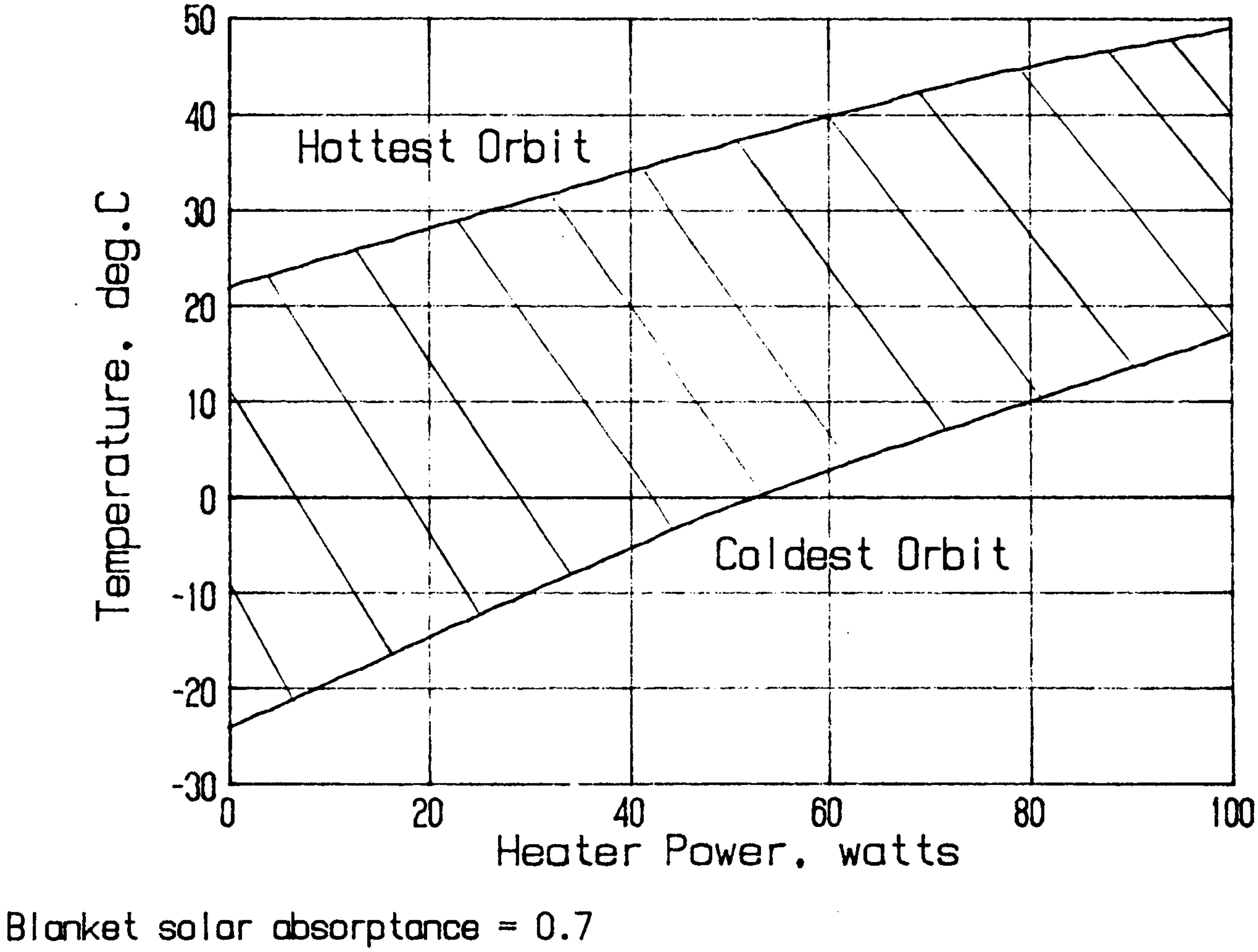


Figure 1. Temperature Range of Blanketed PCC

JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM

3546-SP100-86-003

April 30, 1986

TO: Len Jaffe

FROM: Cathy Cagle CC

SUBJECT: SP100 Space-Based Radar in Orbital Storage

INTRODUCTION

This memo presents the results of thermal analyses of the power conditioning module and its radiators in extended orbital storage for the SP100 Space-Based Radar (SBR) mission.

The configurations studied include the power conditioning module, the shunt and electronics radiators, and the radar antenna at a "storage" altitude of 450 km, and an "operational" altitude of 1088 km.

STORAGE ORBIT

Assumptions: The SBR antenna, power conditioning module and space reactor power system (SRPS) are brought to a 450 km circular orbit by the Shuttle and assembled, where they remain in orbital storage until the Shuttle returns with the Centaur and propellant. It was assumed that during this period there is no attitude control and the system is tumbling as it orbits the Earth. The antenna is stowed and the ends of the exposed power conditioning module are blanketed with multilayer insulation (MLI), as shown in Figure 1.

Two cases were studied; the shunt and electronics radiators were modeled as both blanketed and unblanketed with MLI while in orbital storage. Because the orientation with respect to the Earth and sun is randomly changing, a worst case cold orbit and a worst case hot orbit were studied for each case to bracket the predicted temperatures.

The major assumptions made in this analysis are:

1. Orbital altitude = 450 km.
2. Shunt and electronics radiators modeled as a single black radiator, $\epsilon_{IR} = \alpha_s = .85$
3. Effective emissivity through MLI, ϵ_{eff}, is .02
4. Antenna and SRPS stowed
5. No attitude control (tumbling)
6. Shunt radiator sized for a 300 kW configuration

<u>Results:</u> Figure 2 shows the bracketed radiator temperatures for a blanketed and unblanketed radiator as a function of heater power added to the radiator. It can be seen that blanketing the radiator insulates it from changing surroundings and the difference between the hottest conditions and the coldest conditions is significantly smaller than for the unblanketed radiator.

The results also show that adding as much as 300 watts in heater power does not significantly affect the unblanketed radiator temperature because the heat is radiated into space. The blanketed radiator, however, can be more effectively controlled with a heater and it is shown that approximately 100 to 160 watts, depending on the desired radiator temperature, is required.

OPERATIONAL ORBIT

<u>Assumptions:</u> After the Centaur is attached to the system and the propellant is loaded, the assembly is boosted to an operational orbit of 1088 km. Before the SRPS radiator is deployed and the system is powered, the Centaur is released and the SBR antenna is deployed. Figure 3 illustrates this configuration.

It was assumed that there is attitude control in this orbit, and the system is nadir-pointed. The MLI on the radiators is removed before the system is boosted to 1088km. The SBR antenna is deployed and two cases were considered: the SBR antenna's solar absorptivity was assumed to be both .8 and .2 . The major assumptions are summarized:

1. Orbital altitude = 1088 km.
2. Shunt and electronics radiators modeled as a single black radiator, $\epsilon_{IR} = \alpha_s = .85$
3. Effective emissivity through MLI, ϵ_{eff}, is .02
4. SRPS stowed
5. Attitude control; nadir-pointed
6. Shunt radiator sized for a 300 kW configuration

<u>Results:</u> Figure 4 shows the results of changing the SBR antenna's solar absorptivity. The higher α_s results in a warmer antenna, which increases the loads to the radiators. The 100 to 160 watts of heater power required in the 450 km orbit provides adequate heating in the 1088 km orbit if the antenna's $\alpha_s = .8$. If the antenna's $\alpha_s = .2$, at least 300 watts are required to maintain temperatures above 0^oC.

CONCLUSIONS/RECOMMENDATIONS

It is recommended that the shunt and electronics radiators be insulated with MLI while in the 450 km storage orbit and approximately 100 to 160 watts of heater power used to maintain temperatures above 0^oC. Radioisotope heater units (RHUs) are well suited for this application and should be considered in future studies. After loading the propellant the MLI will be removed and the system boosted into the operational orbit of 1088 km.

The SBR antenna exchanges energy with the radiators, and the use of a high solar absorptivity surface coating on the antenna should be encouraged. This will increase the antenna temperature and the load to the radiators and thus keep the required heater power to a minimum.

It is suggested that the possibilites of decreasing the operating and non-operating temperatures for the electronics, batteries, etc. be explored, as lower qualification temperatures result in lower heater power requirements.

Distribution:

P. Bhandari
E. Chow
R. Ewell
T. Fujita
R. Manvi
W. Menard/B. Wada
B. Nesmith
J. Stallkamp
J. Stevens
J. Stultz

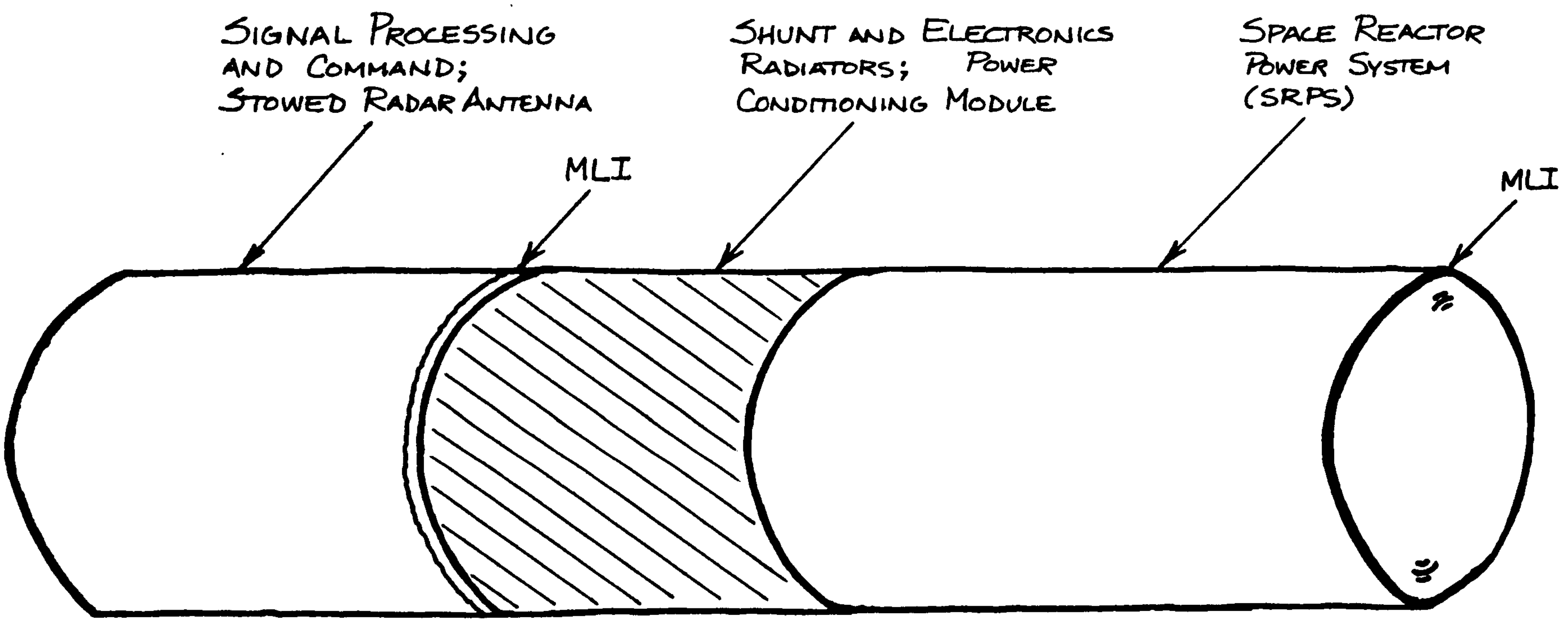


Figure 1. SP-100 SBR Mission: Sketch of Thermal Model in 450 km Orbit

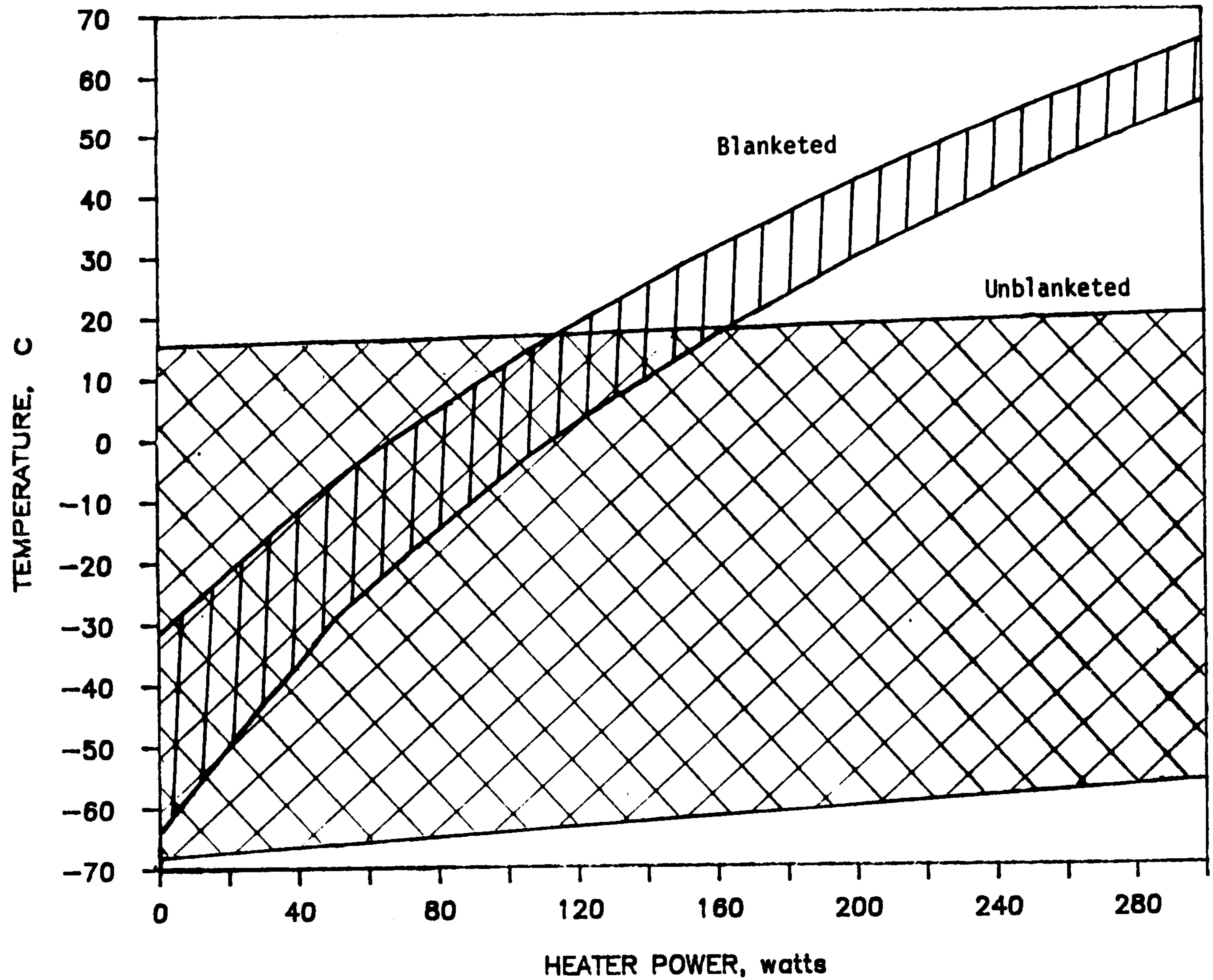
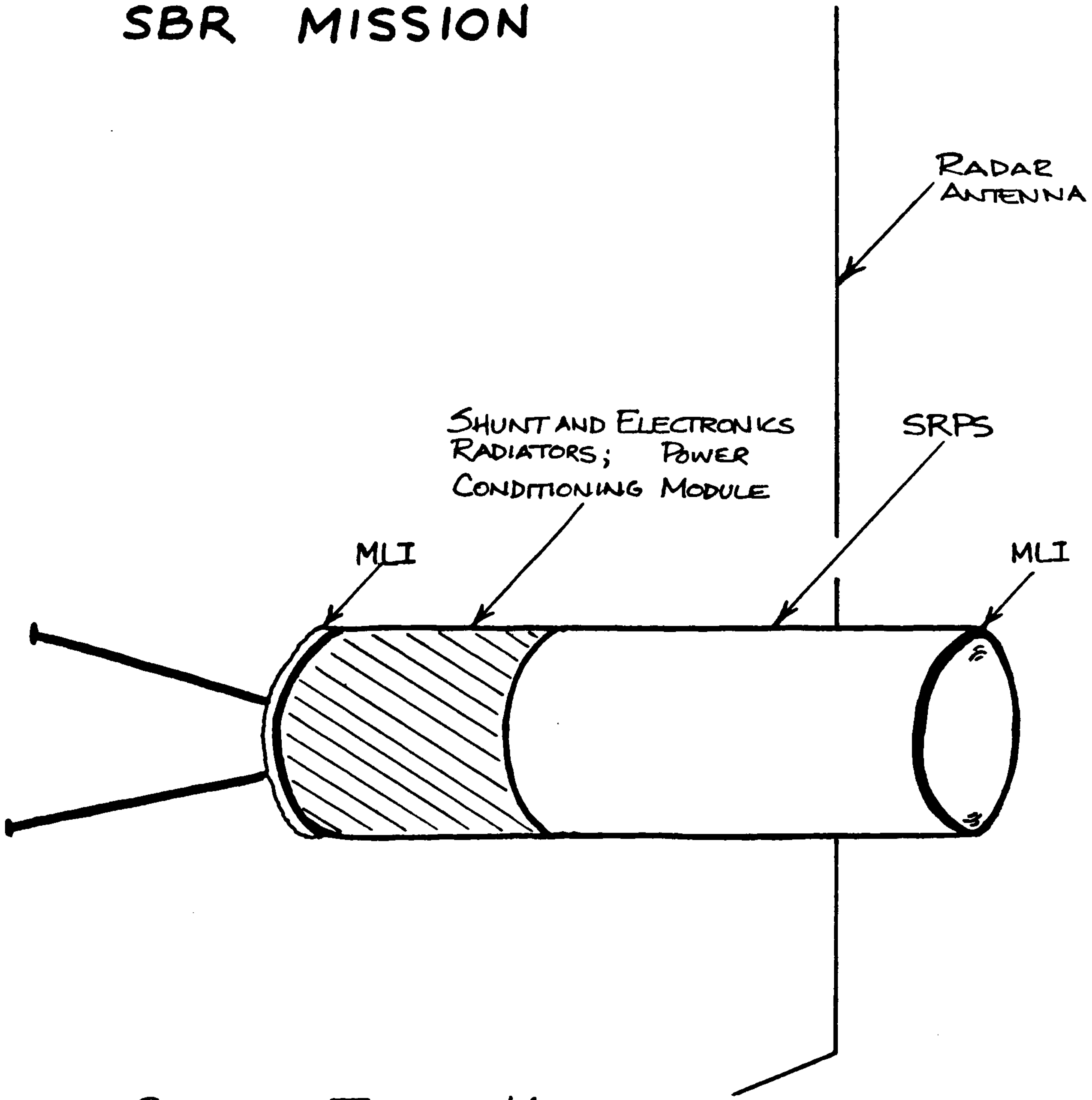


Figure 2. PCM Radiators, 450 km Storage Orbit

SP-100

SBR MISSION



SKETCH OF THERMAL MODEL
IN 1088 KM ORBIT

Figure 3

PCM RADIATORS

1088 km OPERATIONAL ORBIT

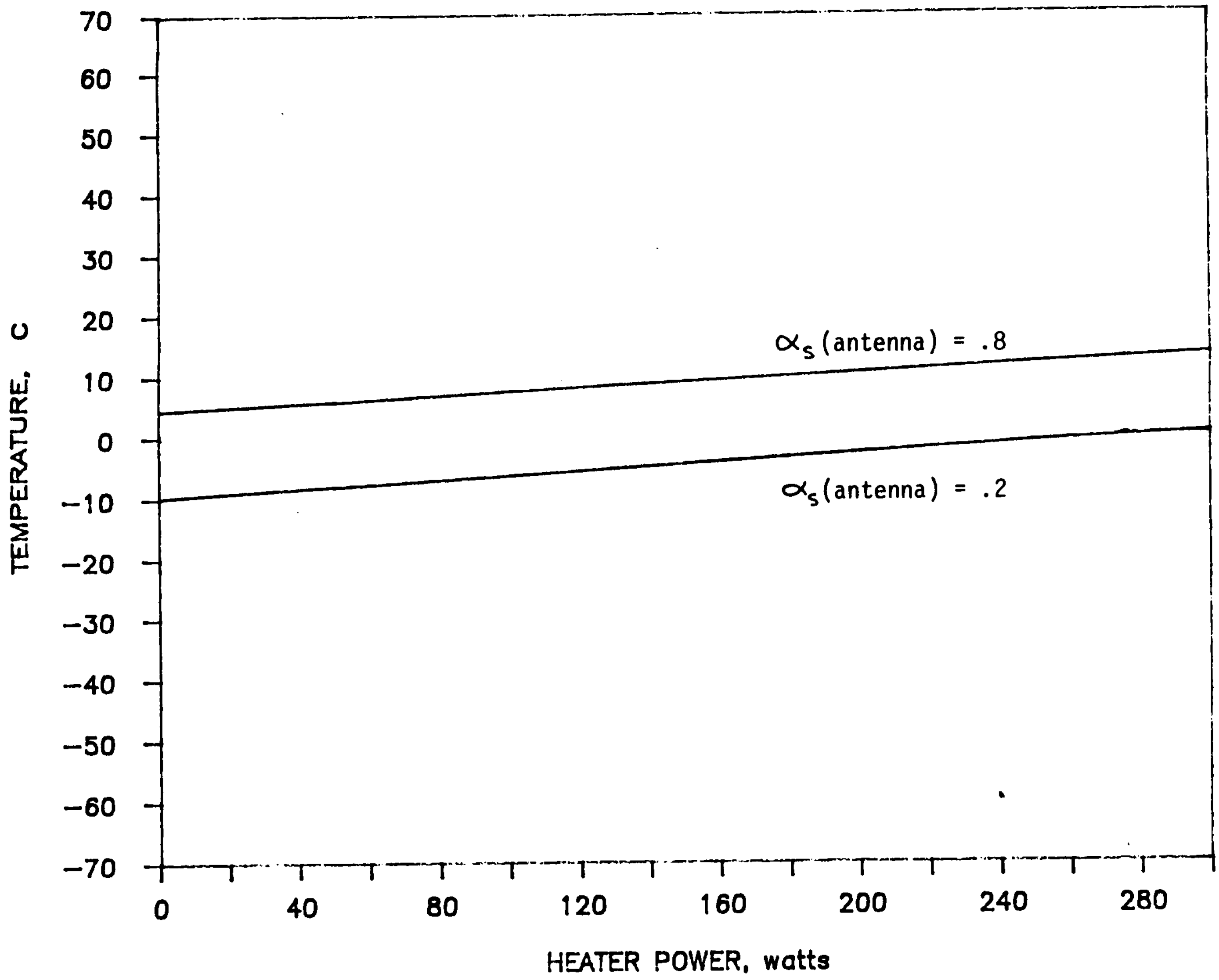


Figure 4

JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM
#342-86-E-068

TO: L. Jaffe

July 7, 1986

FROM: J. Stallkamp

SUBJECT: On-Orbit Temperature Limits for Electronic Parts

MIL-spec rated electronic parts are routinely tested for operating parameters at -55°C. In individual cases turn-on transients may have to be controlled and circuits may not perform at the extreme temperature as specified for use on the normal temperature ranges.

Typical JPL Type Approval temperature limits are -25°C to +75°C, and typical Flight Acceptance values are 0° to 55°C.

It is concluded that on-orbit storage at temperatures between -25°C and 0°C is not likely to present significant problems.

JS:eh

cc: J. Klein

B-10. ATTITUDE CONTROL

T. Kia
J.T. Spanes

February-August 1986

Sections of the attached documents that do not pertain to the radar mission have been omitted.

INTEROFFICE MEMORANDUM

343 - 86 - 1170

Aug. 12, 1986

TO: L. Jaffe

FROM: T. Kia

SUBJECT: SP-100 Orientation using Passive Gravity Gradient Stabilization, revisited

REFERENCES: Kia, T. "sp-100 Orientation Using Passive Gravity Gradient Stabilization," IOM 343 -86- 1129, Aug. 6,1986

This is a revision to the referenced memorandum. The reported moments of inertias for the SP-100/SBR are erroneous. The corrected value are given in the table below.

	I_{xx} (Kg m^2)	I_{yy} (Kg m^2)	I_{zz} (Kg m^2)
OTV	3.86×10^5	6.79×10^6	6.96×10^6
SBR	1.83×10^7	1.16×10^7	2.27×10^7

As a result the SP-100/SBR preferred orientation under gravity gradient will change as follows:

a) The boom axis will be along the velocity vector. That is tangent to the orbit.

b) The long axis of the antenna will be parallel to the nadir direction.

Since this is a null gravity gradient orientation, it will cause only a minor attitude control problem. An active control system can keep the spacecraft in the desired orientation. The desired orientation is defined as,

a) Plane of the antenna perpendicular to the nadir direction,

b) and the plane of the radiator in the orbit plane.

The desired orientation is essentially the orientation with *minimum drag, while the preferred orientation is the one with maximum drag. This will effect among other things the orbital life time, with no active control, drastically. The desired orientation may be achieved by extending the boom. The approximate moment of inertias for different size booms are tabulated below. It is clear from this table that a 40 meter boom will provide the desired stable orientation using passive gravity gradient. Longer boom may also cause lower frequency modes which should be addressed before a *decision concerning the boom length is finalized.

SBR BOOM	I_{xx} (Kg m^2)	I_{yy} (Kg m^2)	I_{zz} (Kg m^2)
35 m	1.83×10^7	1.72×10^7	2.82×10^7
40 m	1.83×10^7	2.05×10^7	3.16×10^7

DISTRIBUTION:

G. M. Burdick
D. G. Carpenter
M. Namiri
H. Otake
J. R. Rose

ATTACHMENT I

Moment of Inertias for the SBR

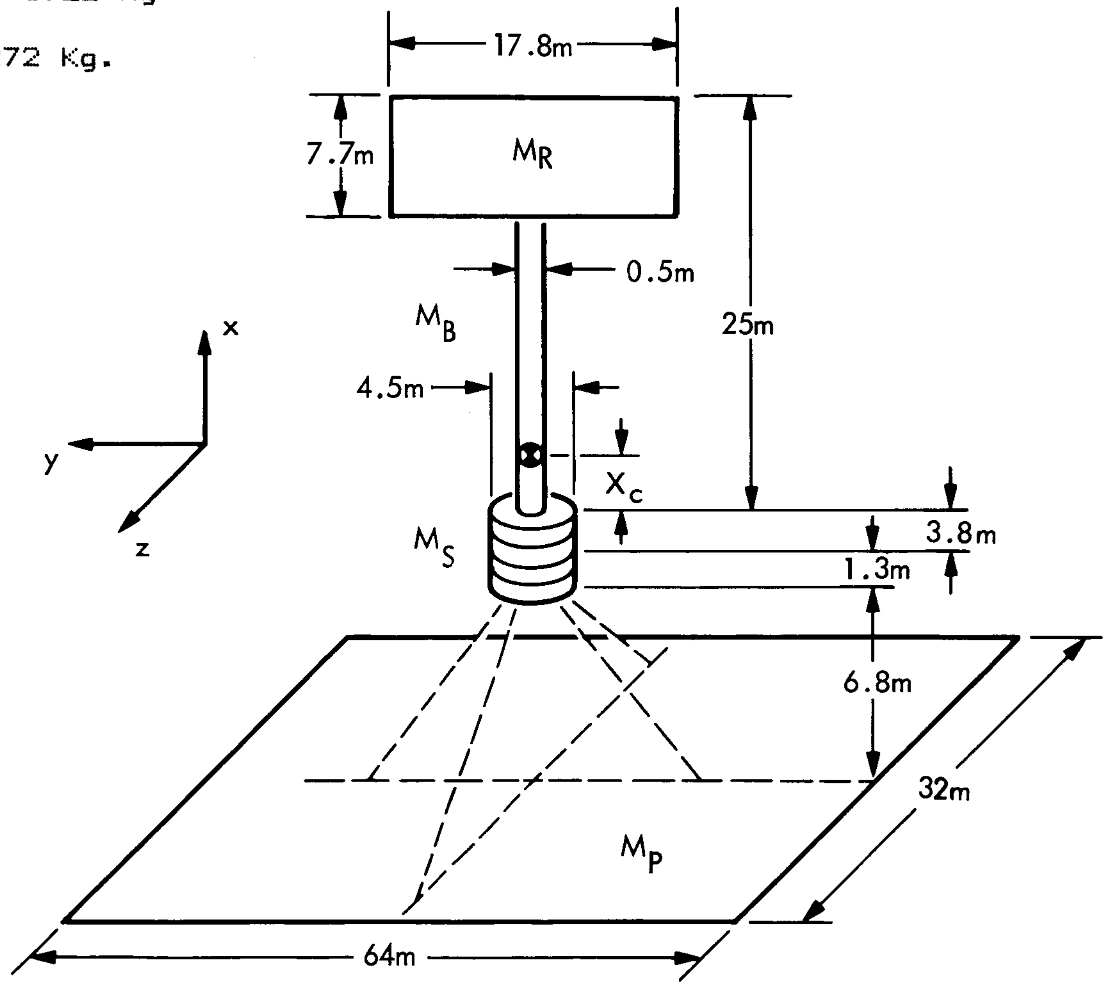
M_R = 8774 Kg.

M_B = 9X(25 - 7.7) = 155.7

M_S = 700+250+250+522

= 1722 Kg.

M_P = 42272 Kg.



CENTER OF MASS

$M_R(25-7.7/2-X_C) = M_S(X_C+(3.8+1.3)/2)+M_P(6.8+3.8+1.4+X_C)$

$X_C[M_S+M_P+M_R] = (25-3.85)\ M_R - 2.55\ M_S - 11.9\ M_P$

$X_C = -\ 6.0995$ meters

INERTIAS

$I_{XX} = I_{RX} + I_{BX} + I_{SX} + I_{PX}$

$= M_R(17.8)^2/12 + M_B(25-7.7)^2/3$

$+ M_S(4.5)^2/2 + M_P(32^2 + 64^2)/12$

$= 1.828761 \times 10^7$ Kg. m^2

$I_{YY} = I_{RY} + I_{BY} + I_{SY} + I_{PY}$

$= M_R[7.7^2/12 + (25-3.85+6.0995)^2]$

$+ M_B[(25-7.7)^2 + 6.1^2]$

$+ M_S[(4.5/2)^2/4 + 5.1^2/3 + 0.9^2]$

$+ M_P[32^2/12 + (6.8+1.3+3.8-6.1)^2]$

$= 1.16628 \times 10^7$ Kg. m^2

$I_{ZZ} = I_{RZ} + I_{BZ} + I_{SZ} + I_{PZ}$

$= M_R[(17.8^2+7.7^2)/12 + 27.25^2]$

$+ M_B[(25-7.7)^2/3 + 6.1^2]$

$+ M_S[(4.5/2)^2/4 + 5.1^2/3 + 0.9^2]$

$+ M_P(5.8^2 + 64^2/12)$

$= 2.27 \times 10^7$ Kg. m^2

SP-100 ATTITUDE CONTROL

J. T. SPANOS

SYSTEMS DESIGN AND
INTEGRATION STATUS
REVIEW

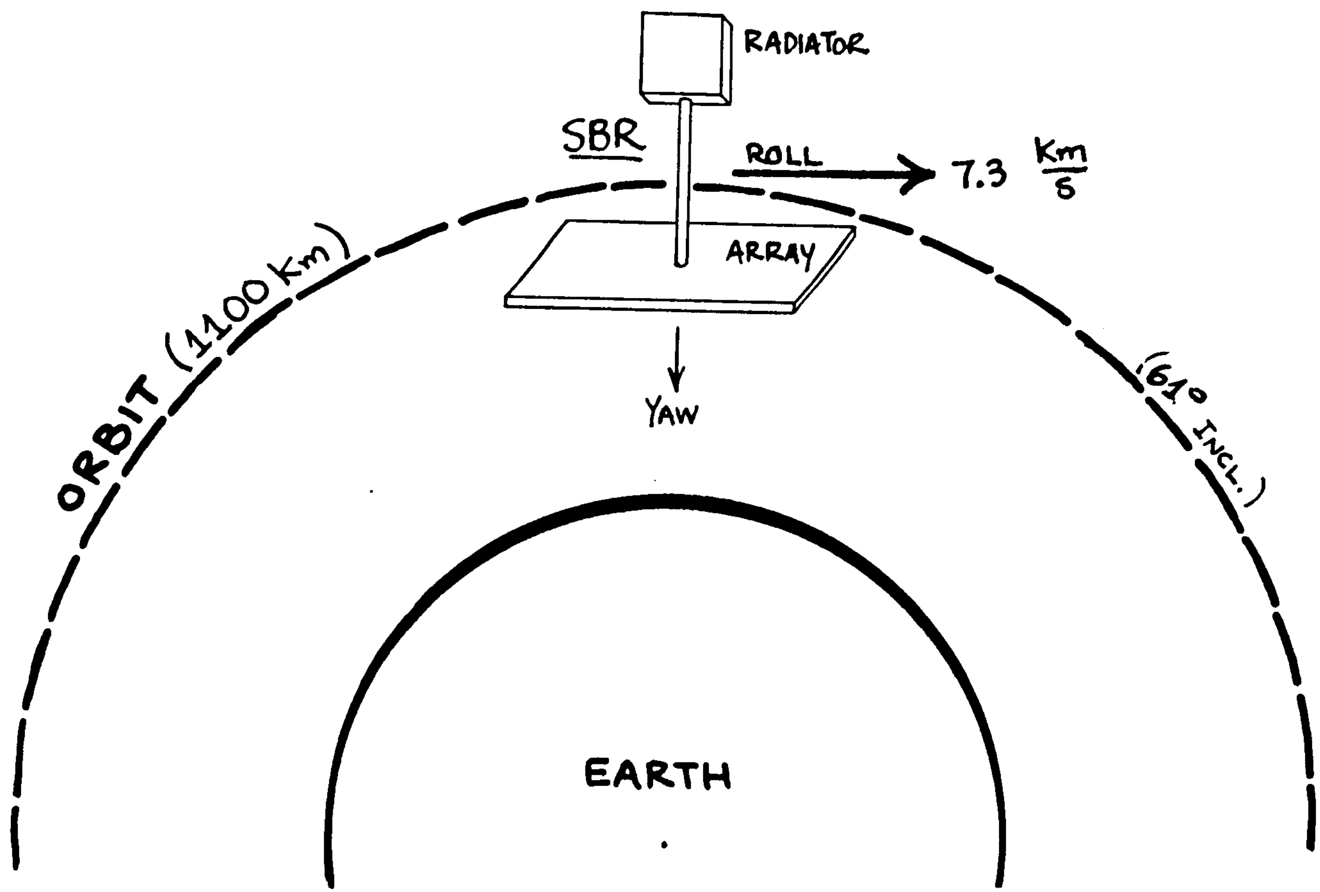
FEBRUARY 11, 1986

AGENDA

- o MISSION REQUIREMENTS
- o DISTURBANCE ENVIRONMENT
- o STRUCTURAL FLEXIBILITY
- o A/C ACTUATION AND SENSING
- o PLANS

SPACED BASED RADAR (SBR)

- o POINTING ACCURACY: ± Ø.2 DEGREES
- o TRACKING: YAW ANGLE FOLLOWS A 3.5 DEGREE SINEWAVE OF PERIOD EQUAL TO THE ORBITAL PERIOD.
- o LOWEST STRUCTURAL FREQUENCY: Ø.Ø1 HERTZ



DISTURBANCE ENVIRONMENT

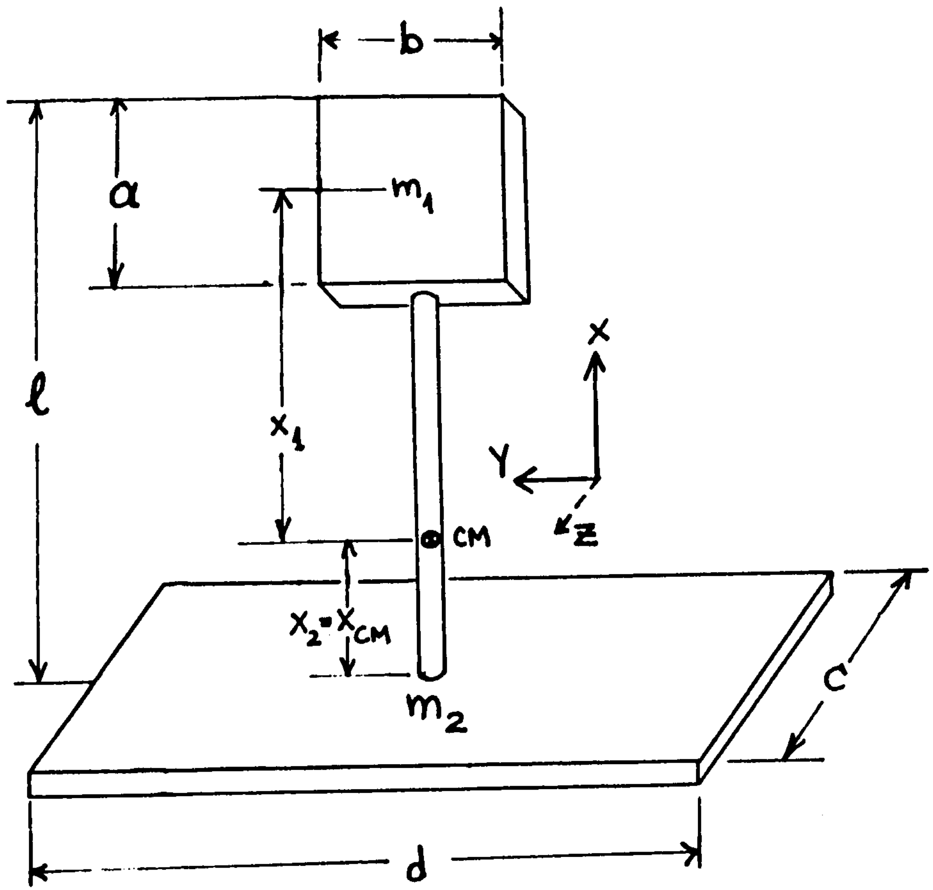
A. EXTERNAL

- o AERODYNAMIC DRAG
- o GRAVITY GRADIENT
- o MAGNETIC FIELD
- o SOLAR PRESSURE

B. INTERNAL

- o STRUCTURAL FLEXIBILITY
- o PROPULSION MASS LOSS
- o FUEL SLOSH
- o THRUST MISALIGNMENTS
- o ON-BOARD MOVING PARTS

GRAVITY GRADIENT TORQUES (SBR)



$$x_{CM} = \frac{m_1}{m_1+m_2}\left(l - \frac{a}{2}\right)$$

$$x_1 = l - \frac{a}{2} - x_{CM}$$

$$x_2 = x_{CM}$$

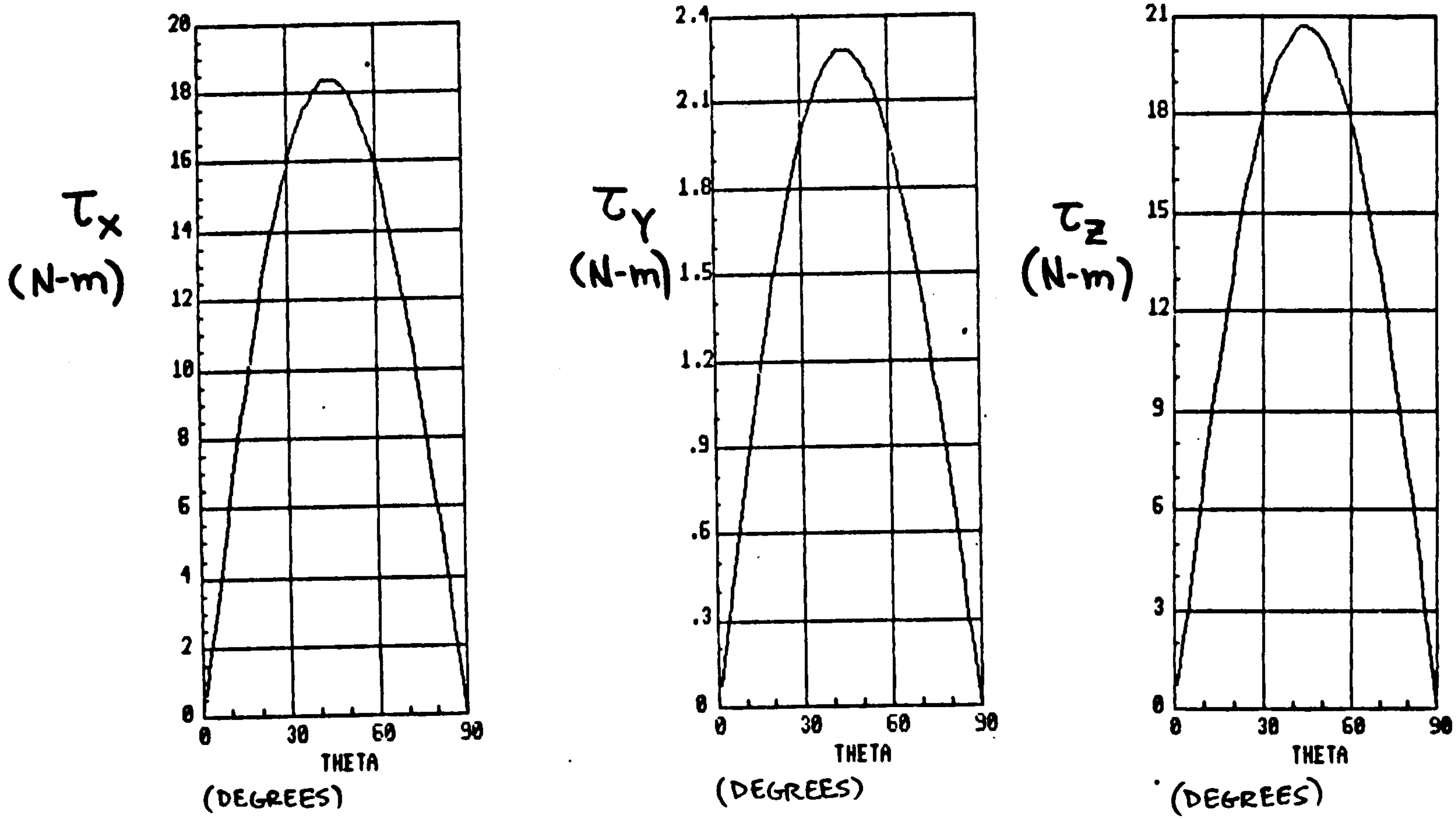
$$\begin{cases} I_{xx} = \frac{1}{12}m_1 b^2 + \frac{1}{12}m_2(c^2+d^2) \\ I_{yy} = \frac{1}{12}m_1 a^2 + \frac{1}{12}m_2 c^2 + m_1 x_1^2 + m_2 x_2^2 \\ I_{zz} = \frac{1}{12}m_1(a^2+b^2) + \frac{1}{12}m_2 d^2 + m_1 x_1^2 + m_2 x_2^2 \end{cases}$$

DATA:

$a = 10\,m$ $\quad m_1 = 7500\,kg$
$b = 10\,m$ $\quad m_2 = 50000\,kg$
$c = 32\,m$
$d = 64\,m$
$l = 25\,m$
$R = 1100\,km + R_\oplus$

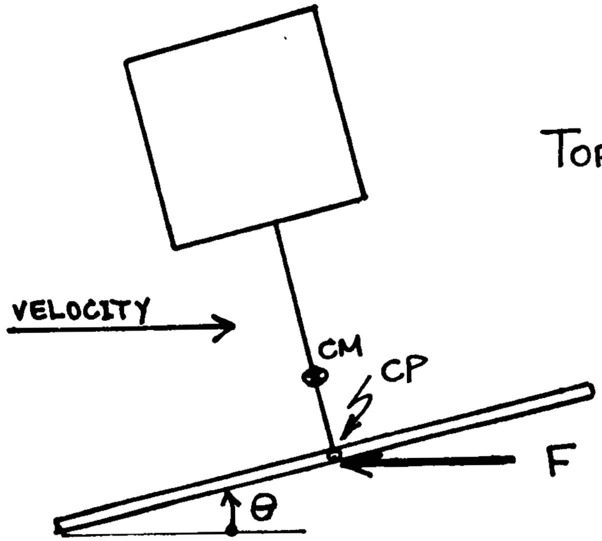
GRAVITY GRADIENT TORQUE: $\frac{3\mu_\oplus}{2R^3}\left(I_{jj} - I_{kk}\right)\sin 2\theta$

GRAVITY GRADIENT TORQUES (SBR)



SOLAR PRESSURE/AERODYNAMIC DRAG (SBR)

(a)

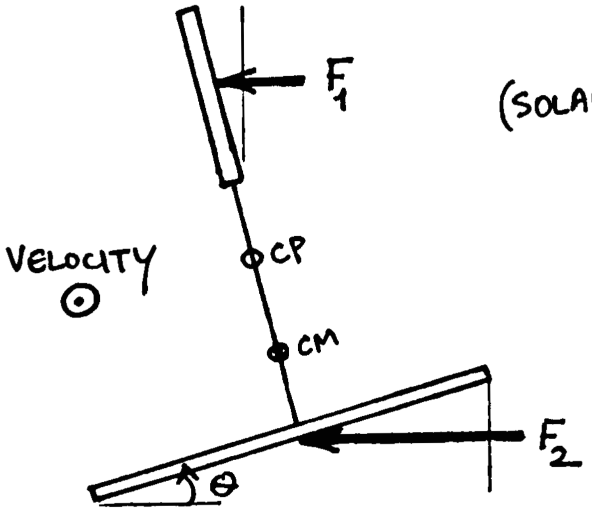


TORQUE : $T = F * X_{CM}\, c\theta$

$F = K * (cd)\, s\theta$

$$\begin{cases} K_S = 4.5 * 10^{-6} \frac{Nt}{m^2} \\ K_D = 2.8 * 10^{-6} \frac{Nt}{m^2} \end{cases}$$

(b)



(SOLAR PRESSURE)

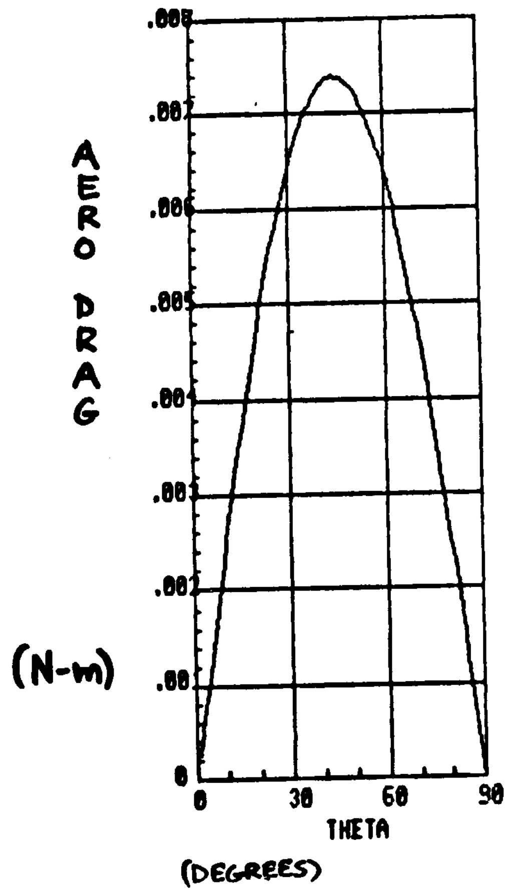
$$X_{cp} = \frac{ab\left(l - \frac{a}{2}\right)}{cd \tan\theta + ab}$$

$$F = F_1 + F_2 = K * [ab\, c\theta + cd\, s\theta]$$

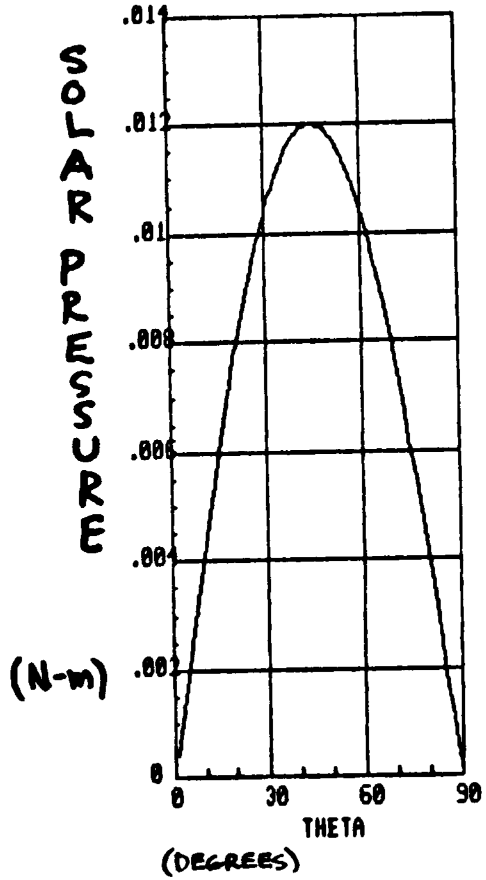
TORQUE : $T = F * (X_{cp} - X_{CM})\, c\theta$

SOLAR PRESSURE/AERODYNAMIC DRAG (SBR)

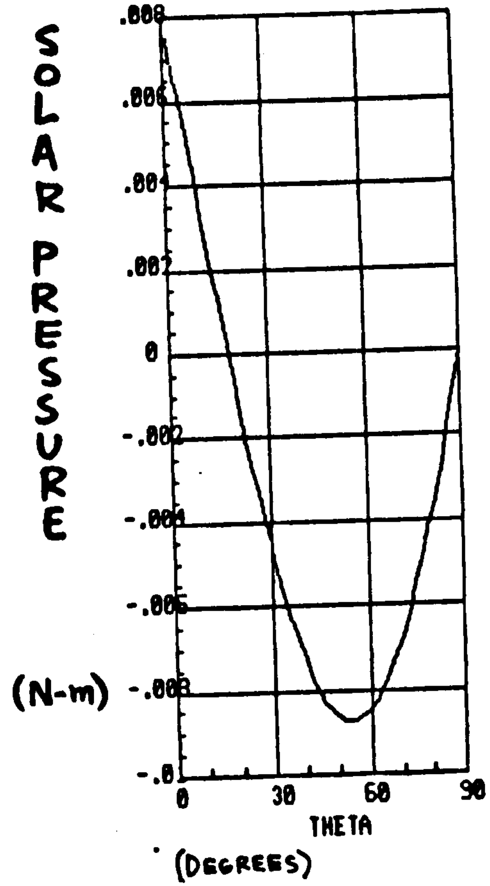
(a)



(a)



(b)



EXTERNAL DISTURBANCE SUMMARY

- THE MOST SIGNIFICANT ENVIRONMENTAL DISTURBANCE TORQUE IS GRAVITY GRADIENT.

- RECOVERY FROM WORST CASE ATTITUDE (I.E. 45^{o}) WILL REQUIRE CONTROL TORQUES IN EXCESS OF 2Ø N-M.

THREE AXIS ACTUATION

o MOMENTUM WHEELS VERSUS ION THRUSTERS

-UNLOADING REQUIRED	-NO UNLOADING
-NO PROPELLANT CONSUMPTION	-CONSUME PROPELLANT

o ACS CONFIGURATION OPTIONS

1. THREE REACTION WHEELS (RW)
   - -FINE CONTROL
   - -LOW TORQUES (1 N-M)
2. THREE CONTROL MOMENT GYROS (CMG)
   - -COARSE CONTROL
   - -HIGH TORQUES (5ØØØ N-M)
3. HYBRID CONTROL (CMG'S & RW'S)
4. THRUSTER SYSTEM (STRATEGICALLY LOCATED AND/OR GIMBALLED)

THREE AXIS SENSING

- o HORIZON SENSOR MEASURING ROLL AND PITCH
- o THREE AXIS INTERTIAL REFERENCE UNIT (GYROS)
- o THIRD AXIS IS CALIBRATED:
  - -THROUGH ORBITAL GYRO-COMPASSING
  - -USING STAR SENSOR

ORIGINAL PAGE IS
OF POOR QUALITY

JET PROPULSION L

INTEROFFICE MEMORANDUM

343 - 86 - 545

Apr. 16, 1986

TO: *L. Jaffe*

FROM: *T. Kia*

SUBJECT: *Initial Concepts for SP-100 Attitude Control on Station*

This note is in response to the Action Item No. 35a. In it a concepts for the on station attitude control of the SP-100 for two possible missions, the Orbital Transfer Vehicle (OTV) and the Space Based Radar (SBR) are presented.

The SBR configuration consists of a large (32m X 64m) radar array attached to the end of long boom. This orbital altitude for this mission is 1100km and the inclination is 61°. The SP-100 reactor and the radiator are attached to the other end of the boom.

OTV on the other hand consists of a cargo bay in place of the radar array. During the normal operations, the OTV will be in transient between two known orbits. But before any given mission it may spend long periods of time in parking, or waiting, orbits. This memo addresses only the attitude operations for those periods.

FUNCTIONAL REQUIREMENTS

The following functions are to be performed by the s/c on the station:

1) *Point payload to any direction to within the required knowledge and stability. SBR payload is nadir pointed. While, on station, OTV payload is assumed to point to nadir.*

2) *Provide 3 axes control on the station.*

3) *Provide antenna pointing towards Earth and/or TDRSS satellite.*

4) *Provide yaw angle tracking capability.*

ON STATION POINTING REQUIREMENT

The tightest pointing requirement is due to the Space Based Radar (SBR). SBR requires pointing accuracy of ±0.2 degrees. The yaw axis should track a 3.5 degree sinewave with a period equal to the orbital period. The stability requirement is TBD.

OTV's on station pointing accuracy on the other hand, is very coarse. On station OTV has two modes of operation, parking and docking. In the parking mode of operation , accuracy requirement is $\pm$ 5 degrees. During the docking operation OTV attitude control system should be capable of maintaining attitude to within $\pm$ 0.5 degrees. There are no requirements on the stability for either modes of operation.

Antenna pointing requirements are TBD.

PROPOSED SYSTEM

A 3-axis stabilized control system is proposed for both missions, Fig. 1. The primary attitude sensor is the three axis Inertial Reference Unit (IRU) package. The IRU package consists of redundant gyros and accelerometers. A horizon sensor may be added for the SBR mission to improve the nadir pointing capability. The primary actuators are three single axis control moment gyros. All of these sensors and actuators are located on the payload side of the s/c. Conventional PID controller will probably suffice for on station requirements of these missions. Moment dumping is performed continuously using the electromagnetic torquers. Figure 2 shows a simplified block diagram for such a closed loop control system.

RATIONAL

A) ACTUATORS

Because of the nature of the SP-100 radiation, all payloads are boom mounted. The boom length is at least 25 meters. Such an structure will tend to have a substantially smaller moment of inertia in the axis along the boom. In orbit, about any planet, this satellite will experience a gravity gradient force which tends to align the boom axis with the nadir vector. This type of stabilization may be sufficient for the OTV in the parking mode, but will not be sufficient for the other OTV mode or for the SBR mission. Actuators are needed to overcome the external torques acting upon the spacecraft, and for pointing and turning, the spacecraft. Either momentum exchange or gas jet thrusters could produce the required torque. Momentum compensation was selected for following reasons,

a) to save on propellant consumption, and
b) to avoid firing of the arc-jet thrusters excessively.

Disturbances torques acting on the spacecraft are as follows:

1) Gravity gradient torque,
2) Solar Pressure,
3) Aerodynamic drag,
4) Internally induced torques such as fuel slosh etc.

The selected actuator should have the capability to overcome all of above disturbances. For the SP-100 configuration under study, gravity gradient produces by far the largest disturbing torque. The gravity gradient torque may be estimated by,

$$T_{GG} = 3 \quad (I_{ii} - I_{jj}) \ Sin(2\ \Theta)/(2\ R^3)$$

where, Θ is the angle between the boom axis and the nadir direction.

Assuming a flat (10m X 10m) radiator panel separated by a 25 meter boom from either the payload, the maximum torque for the OTV and for the SBR mission were calculated, over 9 newton-meter and over 20 newton-meter respectively. Reaction wheel is incapable of producing such a large torque. Control moment gyros (CMG) were selected because they are capable of producing torques in excess of the requirement.

Since the CMG wheels will saturate, momentum dumping will be required. Electric thrusters may be used for this purpose, but for the same two reasons given above, continuous momentum dumping using electromagnetic torquers are recommended. If needed, a current loop along the boom can provide additional torque for dumping the momentum.

B) SENSORS

Gyros in all axis are needed to measure the attitude error rates. Celestial sensors, star and the sun sensors, are required for gyro calibrations to correct for the scale factor and the drift rate, and for the attitude initialization.

A horizon sensor may be added to the SBR mission to improve the nadir pointing accuracy. Addition of such a sensor will remove the orbital position error from the consideration of the total nadir pointing error. In addition the gyros may be calibrated using the horizon sensor, thus eliminating the need for a star sensor.

HIGH ACCURACY CONTROL SYSTEM

For future missions with higher pointing requirements than the ones discussed, the proposed system may be augmented by the addition of an Integrated Platform Pointing and Attitude Control System (IPPACS). Since IPPACS is the baseline control system for the Mariner Mark II spacecraft, it should not require additional development. IPPACS will be mounted on a two degrees of freedom, momentum compensated , inertially stabilized platform, with both axis of revolution through the platform center of mass. Also included on the platform may be an ASTROS star tracker, an ACS processor, and all of the pointed instruments, Figure 3.

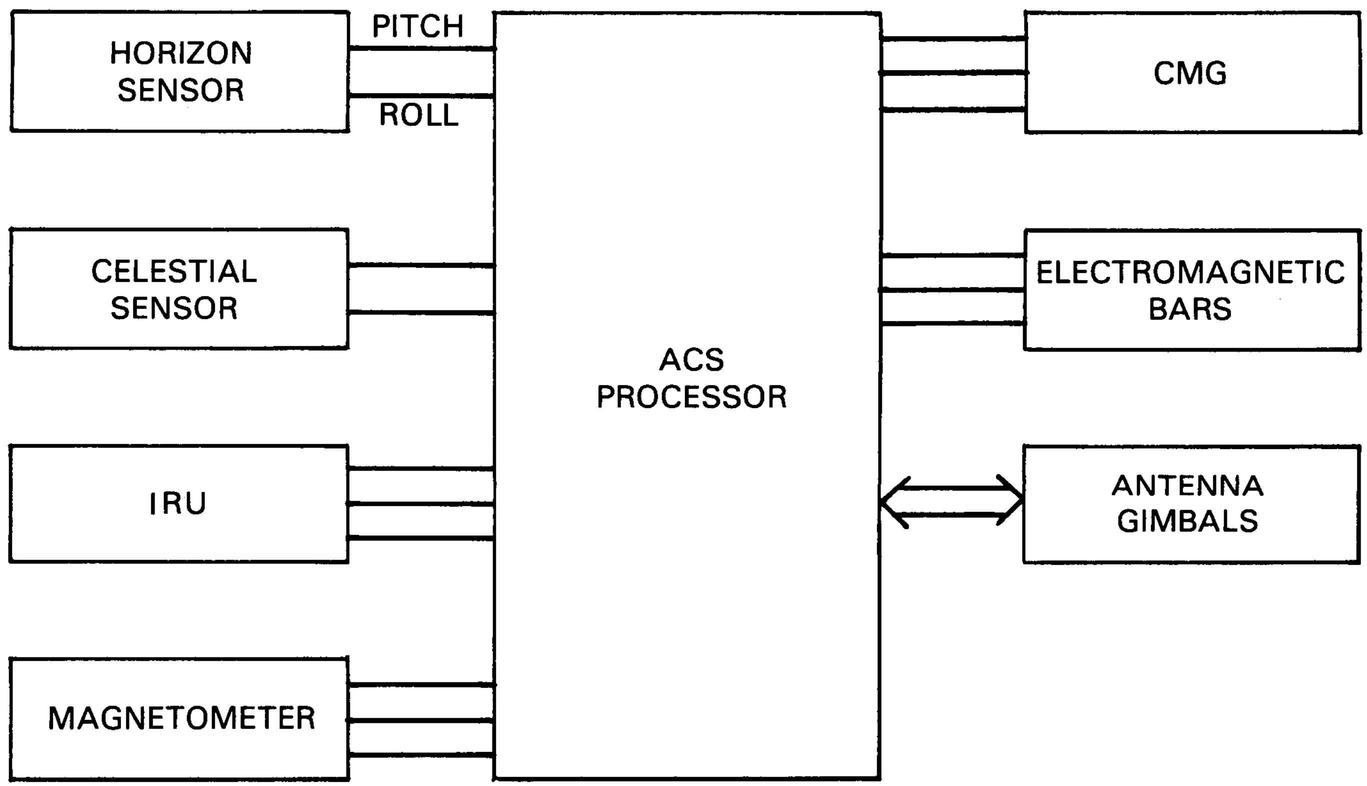


Figure 1. Attitude and Articulation Control System

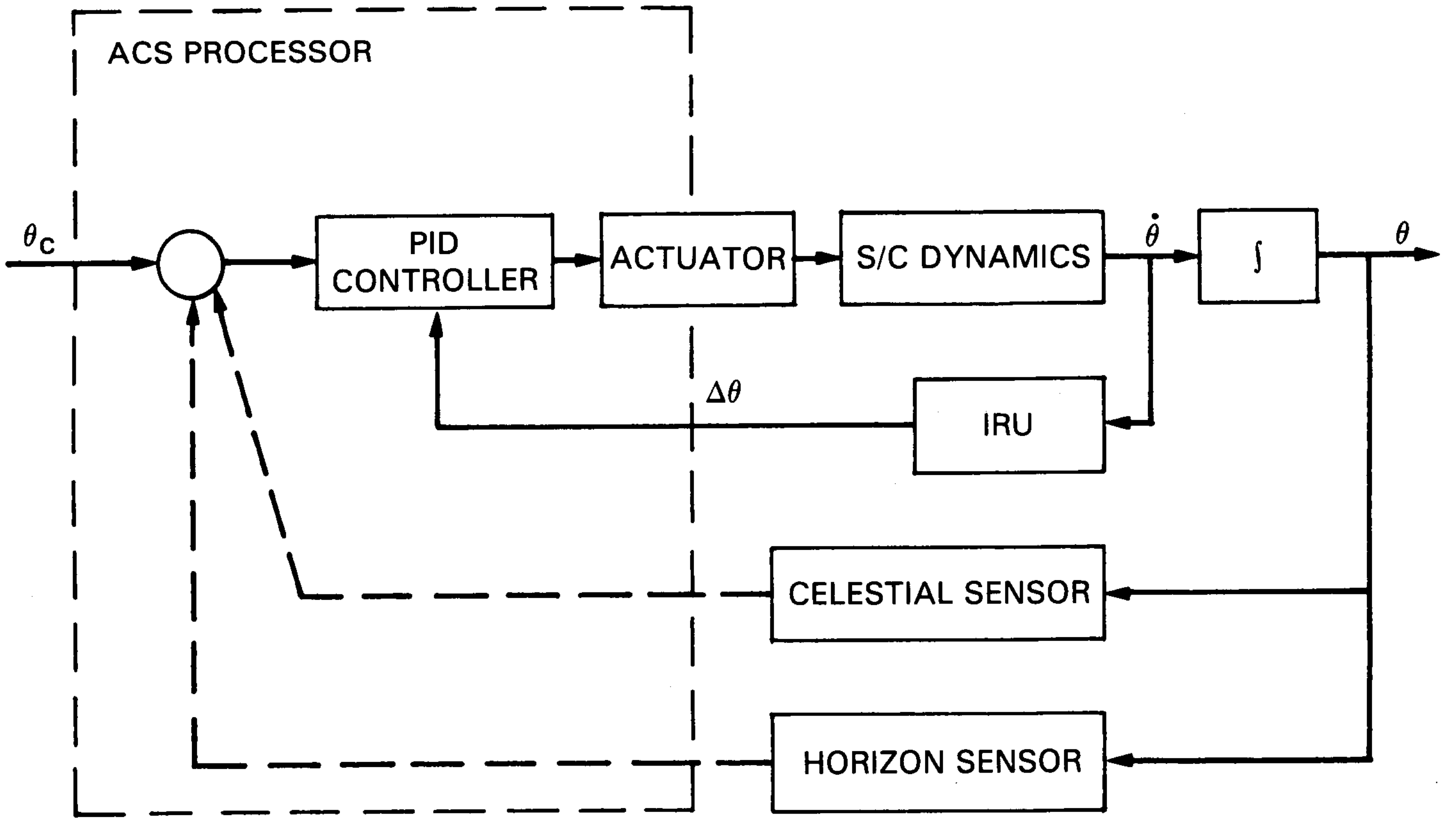


Figure 2. SP-100 Attitude Control Concept

B-11. THERMAL FLUX FROM POWER SYSTEM TO REST OF SPACECRAFT

P. Bhaudari

June 25, 1986

JET PROPULSION LABORATORY | INTEROFFICE MEMORANDUM

3546-TSE-86-079

April 16, 1986

TO: Len Jaffe

FROM: Pradeep Bhandari

SUBJECT: Heat Flux Incident on the Antenna from the SP-100 Shunt Radiator

An analysis has been completed to calculate and plot the heat flux distribution on the antenna due to the SP-100 shunt radiator. Various distances of the base of the radiator from the antenna have been considered for parametric runs.

Figure 1 shows the configuration analyzed. The radiator is 4.3 m in diameter and 1 m in height. The antenna is 64 m long and 32 m wide. The radiator is located symmetrically with respect to the antenna. The range of distances (d) of the base of the radiator from the antenna which have been considered is 5 to 10m.

Figure 2 shows the heat flux distributions as functions of distances from the centerline of the antenna. Each curve represents a constant value of d. Due to cylindrical symmetry, this set of curve suffices to describe the distributions in any radial direction on the antenna. Heat flux is represented as a percentage of the solar constant (1367 w/m^2).

Besides the configuration described earlier, the following assumptions were made for the analysis:

(1) Radiator temperature = 836°K (corresponding to a 300 kw heat output for a surface emittance of 0.8).

(2) Antenna temperature = 313°K (the exact value of this temperature is not very important because of T^4 relationship for heat fluxes)

(3) Heat fluxes presented are the amount <u>incident</u> on unit area of the surface and do not consider the amount which will actually be absorbed by the surface due to its infrared absorptance being less than unity.

(4) Only the heat flux being emitted by the radiator and eventually incident on the surface is considered (i.e., direct or indirect solar radiation, earth & planetary albedo and IR, stellar radiation, etc., are not considered)

<u>Summary of Plots:</u>

(1) In the range of distance, d, considered, the maximum heat flux incident on the antenna is about 23% of the solar constant (SC). The location of this maximum is at a radial distance of 6 m from the center, for a 5 m value of d.

(2) The general trends are:

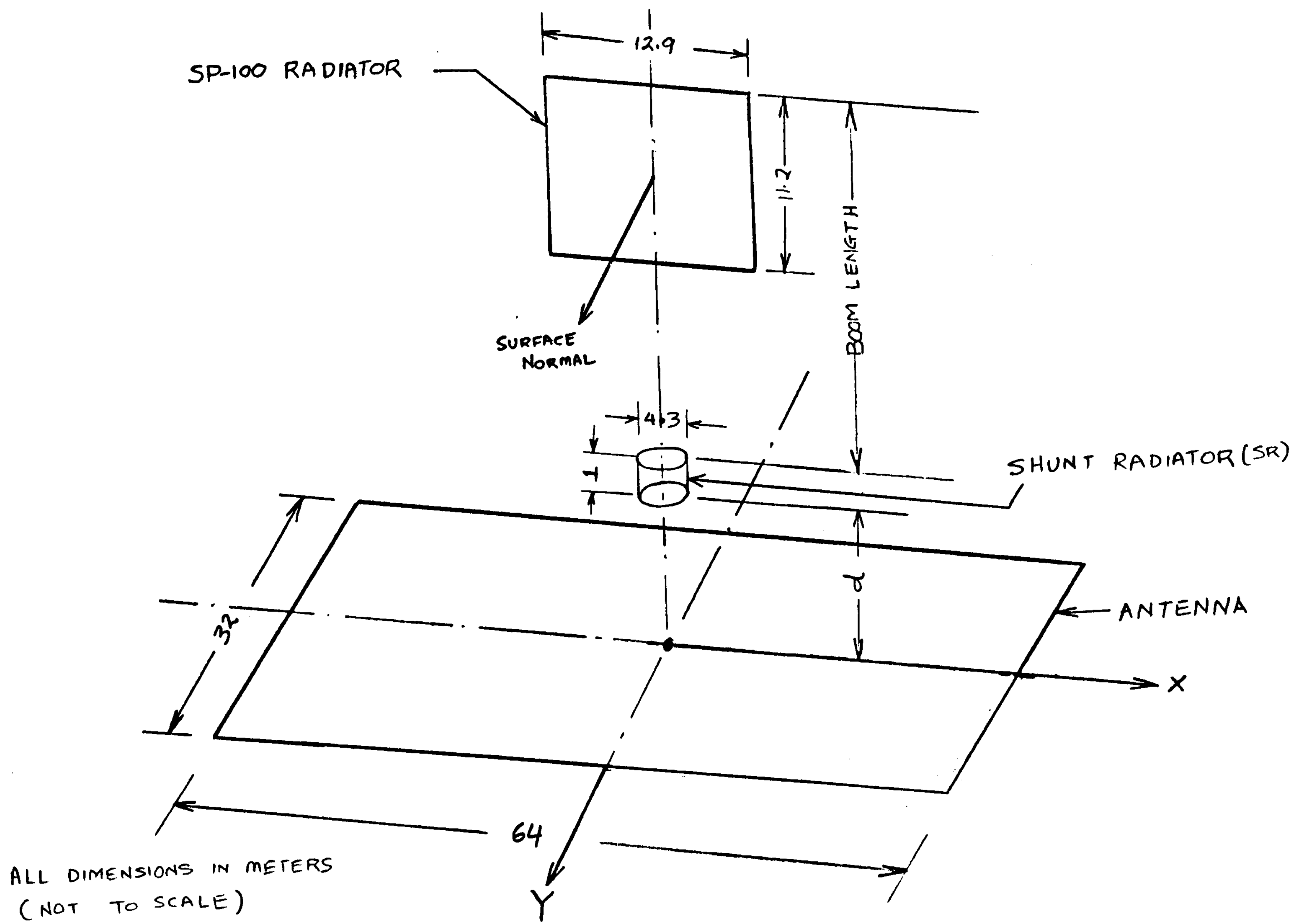
A: For a constant distance of the base of the radiator from the antenna, d, the heat flux vs. distance from the center curve exhibits peaks close to the centerline. The location of this peak shifts farther away from the center as the distance, d, is increased.

B: For any point on the antenna, increasing d decreases the heat flux on the point.

ct
Attachment

<u>Distribution:</u>

Ed Chow	Bill Nesmith
Rich Ewell	J. Stallkamp
Tosh Fujita	Jim Stevens
W. Menard/B. Wada	James Stultz



SP-100 RADIATOR
12.9
11.2
BOOM LENGTH
SURFACE NORMAL
4.3
T
SHUNT RADIATOR (SR)
d
ANTENNA
X
Y
32
64
ALL DIMENSIONS IN METERS
(NOT TO SCALE)

Figure 1

HEAT FLUX ON THE ANTENNA FROM SP100 SHUNT RADIATOR

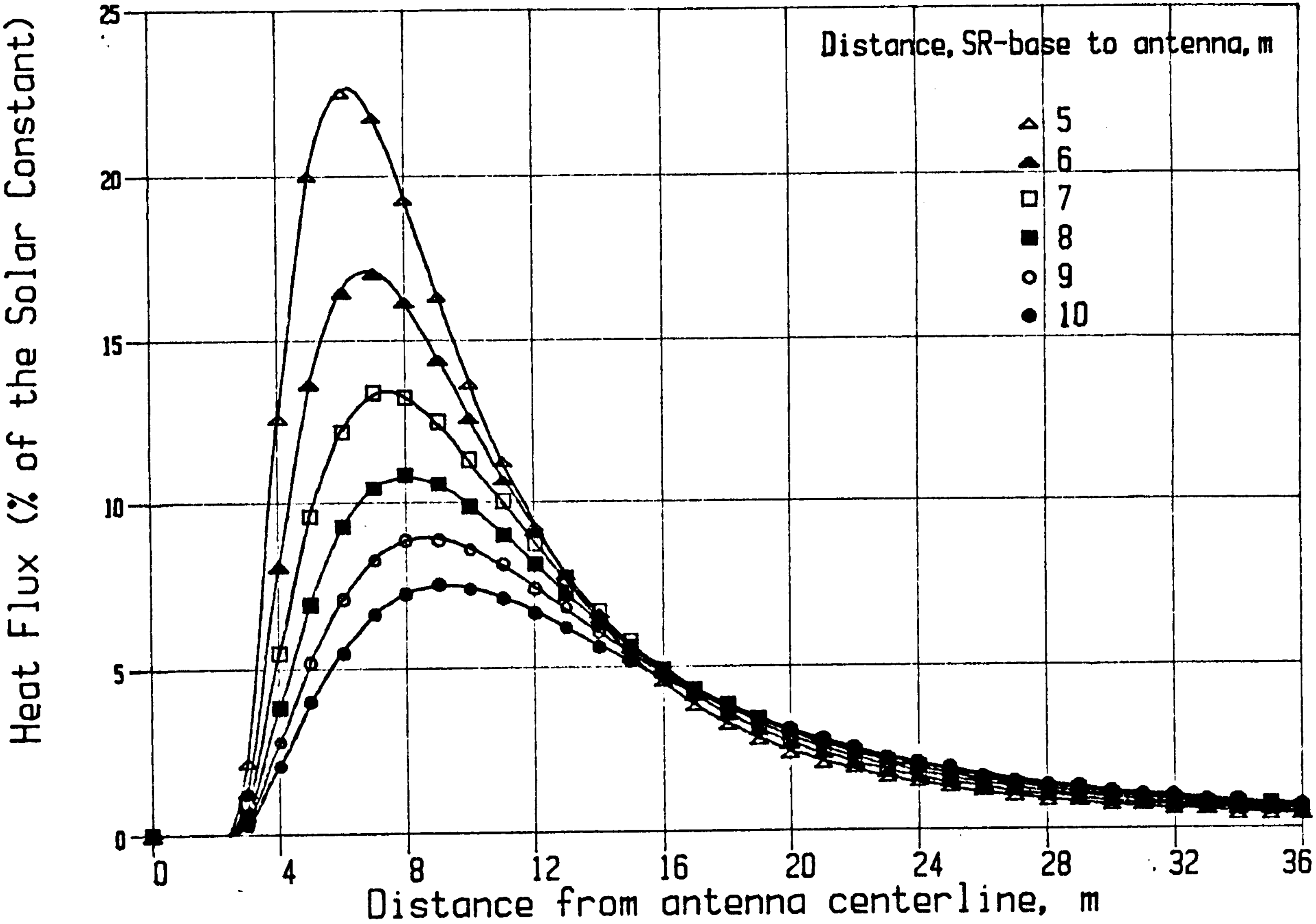


Figure 2

JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM

3546-TSE-86-125

June 25, 1986

TO: Len Jaffe

FROM: Pradeep Bhandari

SUBJECT: Comparison of Various Radiator Configurations in Terms of Heating the SP-100 Radar or User Plane (300 kWe Design)

INTRODUCTION

Various radiator configurations are currently being studied for the SP-100 project. These include the cross, roll-out flat panel, and cone-cylinder design. This memo concentrates on comparing the various concepts in terms of heating the radar (antenna) and the user plane.

RADIATOR CONFIGURATIONS

Previous memos presented by me discussed the flat panel and cone cylinder concepts in detail. Another concept which was studied recently is the cross design. Figure 1 shows in general how the main radiator is located with respect to the shunt radiator, PCC, and the radar antenna. The user plane and the separation distance are defined in the figure.

Figure 2 shows the three radiator design concepts which are compared in this memo. Previous memos on the flat panel radiator concept had assumed a radiator operating temperature of 885°K (supplied by Rich Ewell) and radiator dimensions of 12.9m x 11.2 m (WXH). Recently a new design was proposed with dimensions of 17.8 m x 7.7 m (roll-out flat panel).

The cone-cylinder configuration proposed by GE used a design radiator operating temperature of 836°K. This temperature was used to describe the performance of the cone-cylinder concept in previous memos.

Due to the fourth power law (Figure 3) the effect of radiator temperature on the heat fluxes is very dramatic. Hence, to make a proper comparison the three radiator designs were analyzed on the common bases of temperature (836°K); emissivity (0.8); separation distance (25m, as defined in figure 1); and distance of user plane to radar (6m).

HEAT FLUX ANALYSIS

The cone-cylinder design is axisymmetric. Hence at any radial distance from the centerline of the radar, the heat flux distribution should be independent of direction. The same trend holds for the shunt radiator.

The normal to the roll-out flat plane is along the short dimension of the radar (direction B, Fig. 2). This is the direction along which the peak heat flux is located for this design.

For the cross design, direction A is where the peak heat flux is located. This direction is defined in figure 2, and is at an angle of 45^{o} to the cross panels.

ASSUMPTIONS

(1) Main and shunt radiator temperature = 836^{o}K

(2) ϵ_{ir} = 0.8

(3) Electronics operating temperature = 300^{o}K

(4) Separation distance (reactor to user plane) = 25 m

(5) Distance of user plane to radar = 6 m

(6) Heat fluxes presented are the amount <u>incident</u> on unit area of the surface and do not consider the amount which will actually be absorbed by the surface due to its infrared absorptance being less than unity.

(7) Only the heat flux being emitted by the radiator and eventually incident on the surface is considered (i.e., direct or indirect solar radiation, earth & planetary albedo and IR, stellar radiation, etc., are not considered).

RESULTS

For each configuration, the direction in which the maximum heat flux is located was found, and the heat flux variation (% of solar constant) in that direction was plotted. Figure 4 shows such a plot for the radar plane. The cone-cylinder design shows the highest peak, about 95% of the solar constant. The peak for the cross is about 32% while it is 30% for the roll-out flat plate. The shunt radiator plot shows a peak of about 10%.

Notice the different shapes of the plots for each design. The cone-cylinder exhibits a peak close to the center, at a distance of 5 m. The shunt radiator shows similar trends, with its peak located at about 7 m from the center. The cross and flat plane, however, have peaks located much farther away, at the edge of the radar.

Since the locations of peaks for the cone-cylinder and the shunt radiator are close to each other, they will tend to add up and increase the peak heat flux when both the main cone-cylinder and shunt radiators are operating at full load. This is not the case with the cross and flat plane configuration.

Another aspect to be considered for the cone cylinder case is the large difference between the peak and minimum heat flux. This might require a large difference in the radar coating emittance in order to control the operating temperatures within reasonable limits. Shielding the radiator or radar at strategic points should also be beneficial for this design.

The flat plate roll out panels heat flux distribution is very direction sensitive. The peak flux location is maximum in direction B, perpendicular to the radiator; while in a direction along the radiator plane, the heat flux is zero. This variation in heat flux, however, is not a severe as the cone-cylinder concept. For the cross design, the peak heat flux is located in direction A, at 45^{o} to the cross panels (32% of SC). The maximum heat flux among the panel directions is about 21% of SC, which is not significantly different. Hence the cross is more direction insensitive as compared to the flat panel design.

Figure 5 shows corresponding plots for the heat flux distribution on the user plane. The general trend is that the heat fluxes are much higher than those on the radar plane for the cone-cylinder design (the peak is about 2.5 times as large). The peaks for the cross and flat panel designs are about 1.5 times larger than those on the radar plane. Notice that the peak heat flux due to the shunt radiator is about 7 times larger than on the radar.

CONCLUSIONS

(1) The cone-cylinder design presents the most severe thermal loads on the radar and user planes compared to the cross and flat panel designs.

(2) The cross and flat-panel designs are very comparable in this respect, with the cross imposing slightly higher peak fluxes.

(3) The cone-cylinder design has very large variations in the heat flux (as a function of location). This would require large variations in surface emittances of the radar antenna or strategically located shields.

ct

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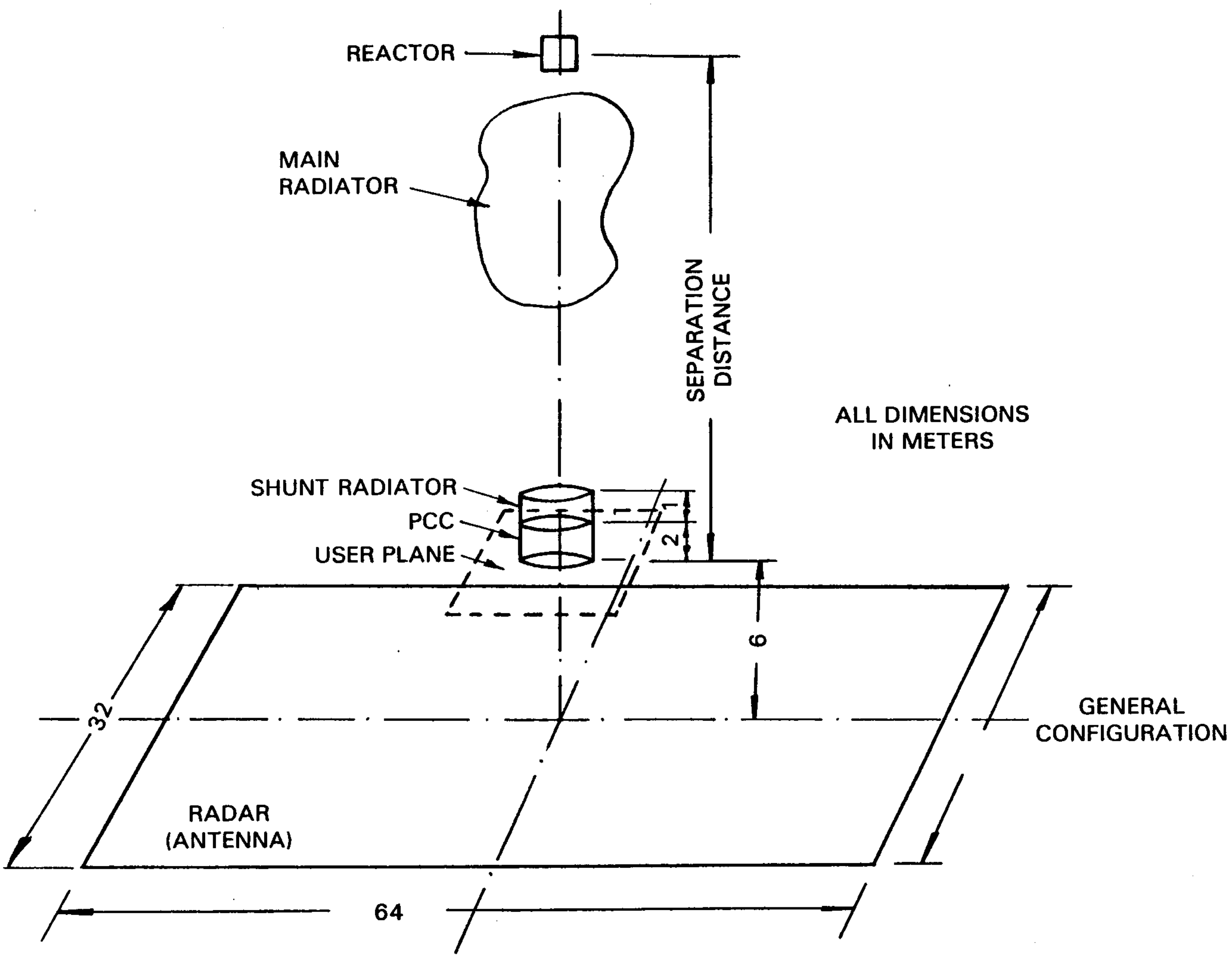
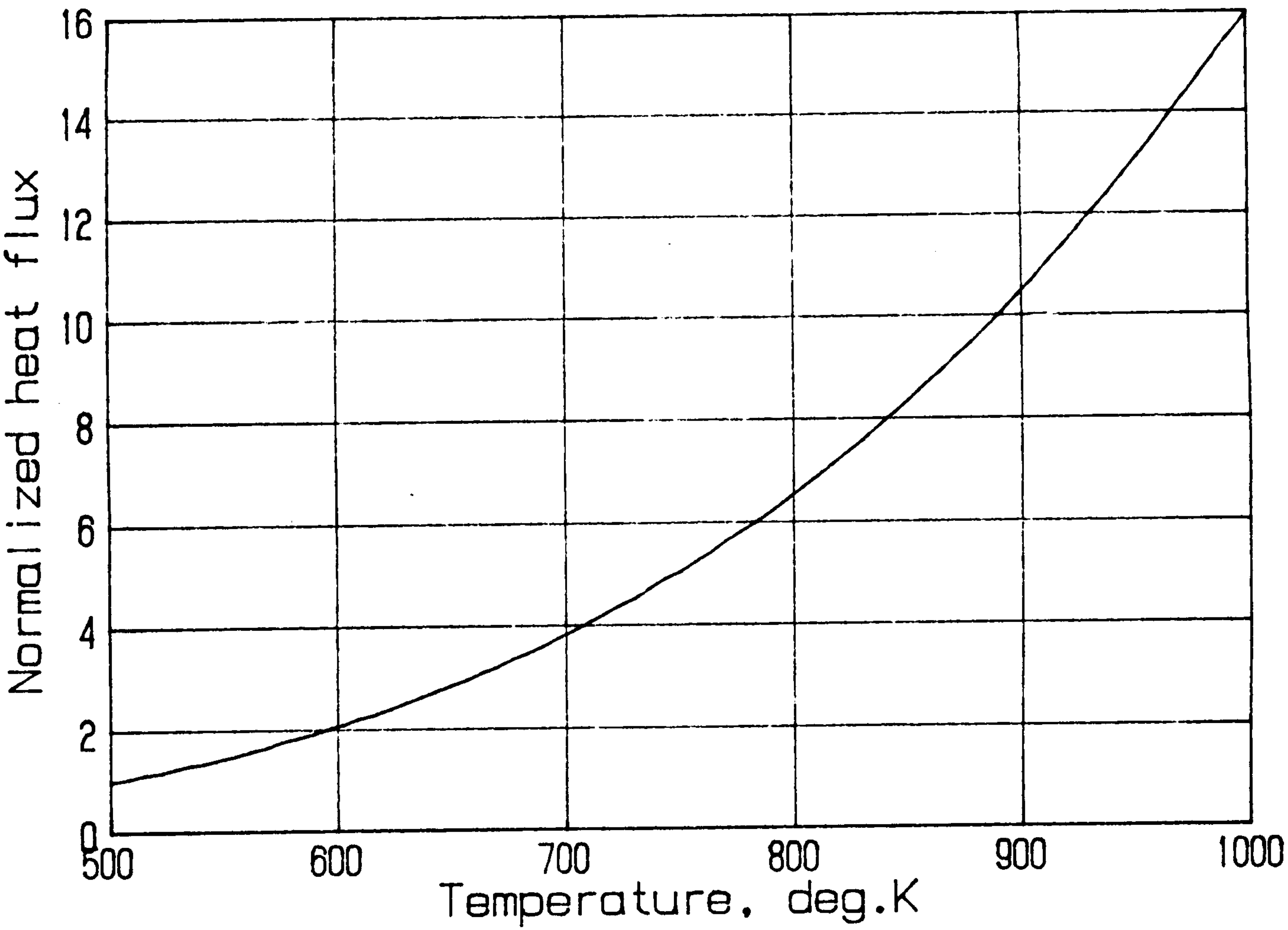

REACTOR
MAIN
RADIATOR
SEPARATION
DISTANCE
ALL DIMENSIONS
IN METERS
SHUNT RADIATOR
PCC
USER PLANE
1
2
6
32
GENERAL
CONFIGURATION
RADAR
(ANTENNA)
64

FIGURE 1

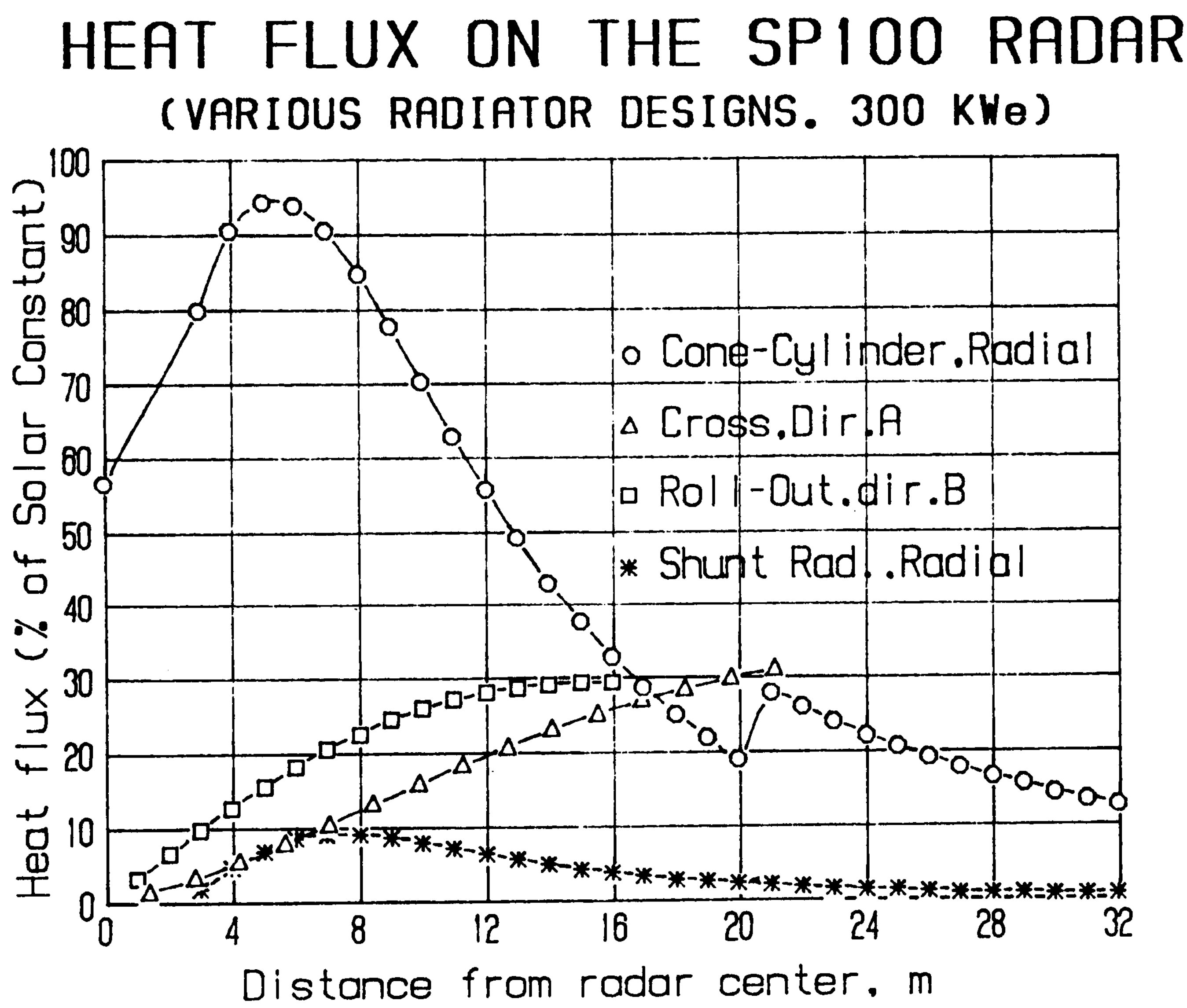
EFFECT OF RADIATOR TEMPERATURE ON HEAT FLUX

(Fourth power law)



500 deg.K reference temperature

Figure 3

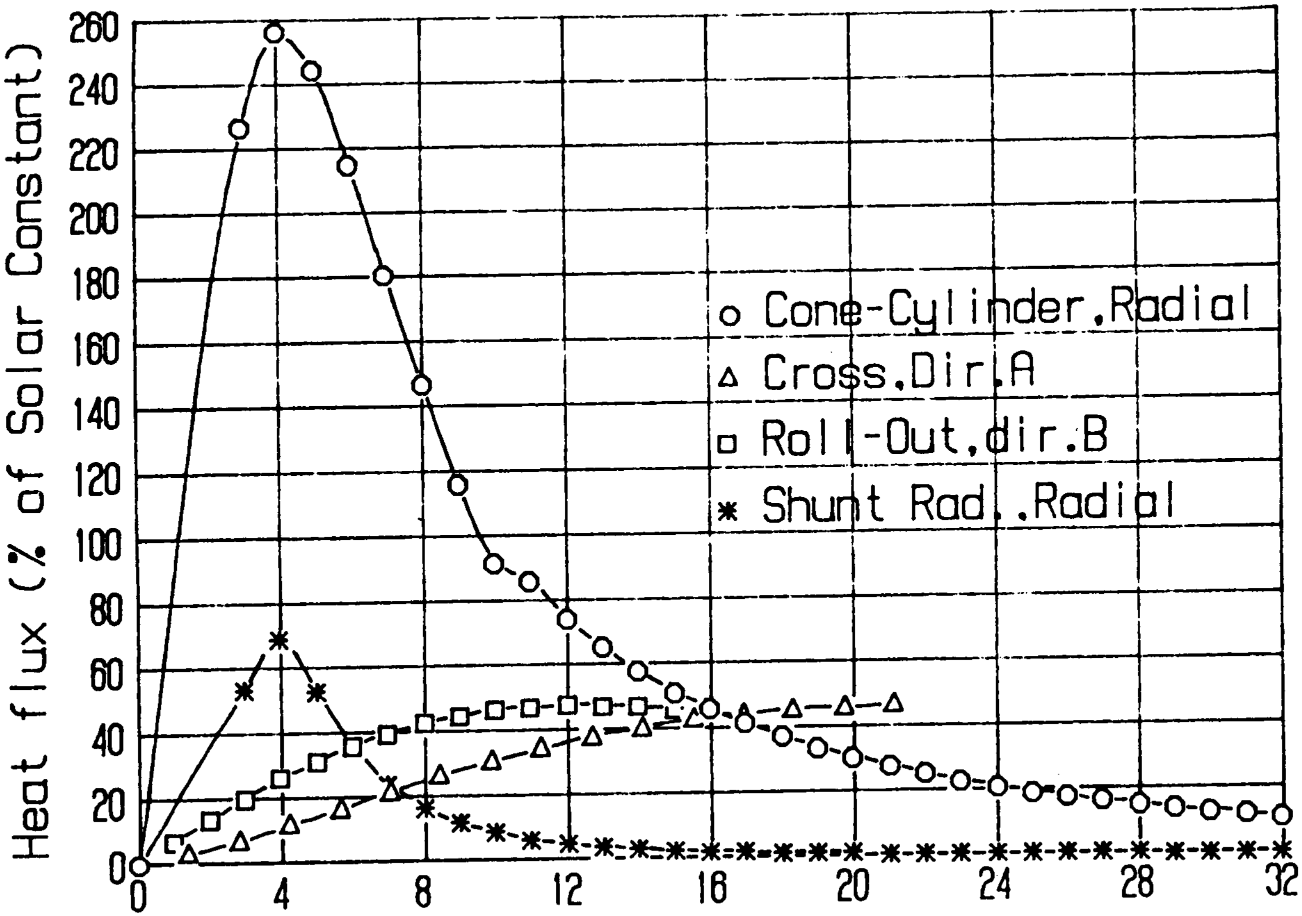


25 meters separation distance

Figure 4

HEAT FLUX ON THE SP100 USER PLANE

(VARIOUS RADIATOR DESIGNS. 300 KWe)



25 meters separation distance

Figure 5

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16. Abstract

A space-based radar mission and spacecraft, using a 300 kWe nuclear reactor power system, has been examined, with emphasis on aspects affecting the power system. The radar antenna is a horizontal planar array, 32 x 64 m. The orbit is at 61 deg, 1,088 km.

The mass of the antenna with support structure is 42,000 kg; of the nuclear reactor power system, 8,300 kg; of the whole spacecraft about 51,000 kg, necessitating multiple launches and orbital assembly. The assembly orbit is at 57 deg, 400 km, high enough to provide the orbital lifetime needed for orbital assembly.

The selected scenario uses six Shuttle launches to bring the spacecraft and a Centaur G upper-stage vehicle to assembly orbit. After assembly, the Centaur places the spacecraft in operational orbit, where it is deployed on radio command, the power system started, and the spacecraft becomes operational. Electric propulsion is an alternative and allows deployment in assembly orbit, but introduces a question of nuclear safety.

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